INTEGRAL LAUNCH AND REENTRY VEHICLE SYSTEM

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VEHICLE PERFORMANCE AND OPERATIONS

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FOREWORD

This volume of McDonnell Douglas Astronautics Company Report Number MDC E0049 constitutes a portion of the final report for the "Integral Launch and Reentry Vehicle Systems Study". The study was conducted by the MDAC for the NASA-Langley Research Center under Contract NAS9-9204.

The final report consists of the following:

Executive Summary

Vol. I - Design, Configuration and Subsystems

Vol. II - Performance, Aerodynamics, Mission and Operations

Vol. III - Plans, Costs, Schedules, Technologies

Vol. IV - One and a Half Stage

McDonnell Douglas Astronautics Company gratefully acknowledges the cooperation of the companies which provided technical assistance during this study. They are:

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ABSTRACT

This study emphasized a two stage to orbit reusable spacecraft system for use in transporting cargo and passengers to and from a near earth orbital space station. A single conceptual "point" design was treated in detail and several alternate systems, corresponding to alternate payloads (size and weight), were examined based on parametric excursions from the "point" design. The overall design goal was to configure the carrier and orbiter vehicles to minimize operational and program recurring costs. This goal was achieved through high system reliability, vehicle recoverability,and rapid ground turnaround capability made possible through modular replaceable component design and use of an integrated onboard self test and check-out system. Launch and land landing of both stages at the ETR launch site was a study groundrule as was the nominal 25,000 lb payload delivered to and returned from orbit and packaged in a 15 ft. diameter by 30 ft. long cylindrical canister. The resulting system has a gross lift-off weight of 3.4 million pounds.

The Orbiter is a 107 ft. HL-10 configuration, modified slightly in the base area to accommodate the two boost engines. The launch propellant tanks are integral with the primary body structure to maximize volume available for propellant.

The Carrier is a 195 ft. clipped delta configuration with ten launch engines identical to those of the orbiter. A dual lobed cylindrical launch propellant tank forms the primary body structure. A 15% thick delta wing is incorporated which contains the landing gear, airbreathing engines and propellant.

A broad range of weight, cost and performance sensitivity data were generated for the baseline and alternate system designs. Pertinent development and resource requirements were identified, development and operational schedules were prepared and corresponding recurring and non-recurring cost data were estimated. Program plans were outlined for the design, manufacture and testing of the Orbiter and Carrier vehicles and for the pursuit of critical technologies pacing vehicle development.

Stage and a half and reusable systems employing expendable launch vehicles were considered initially, but, these efforts were subsequently terminated prior to completion. The expendable launch vehicle data are reported separately. The stage and a half effort employed a version of the McDonnell Douglas Model 176 with four drop tanks.

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1.0 INTRODUCTION

The performance and operational analyses are referenced to the baseline logistics mission requirements. These mission requirements include transportation of 25,000 lb of payload, packaged in a 15 ft. diameter by 30 ft. long cylinder, to a 270 na mi altitude/55° inclination circular orbit, and returning the same payload to earth. The required mission duration is seven days, and a crew of two is required in both the carrier and orbiter. Cruise back to the launch site is provided for the carrier and the orbiter while not having a cruise requirement does have a 10 minute go-around or wave-off requirement.

This volume considers the aerodynamic and thermodynamic performance of the reusable two stage system for each phase of the baseline mission. Launch, entry and special abort trajectories with associated heating analyses, as well as the hypersonic and subsonic aerodynamic requirements and characteristics are discussed. Also included in this volume are (1) the mission analyses, which define the baseline and alternate operational modes and assess the capability of the baseline system to perform alternate missions, and (2) the primary operational analyses which include ground turnaround and cargo handling and crew accommodations. Four special emphasis studies are included in this volume, namely, Abort, Approach and Terminal Landing, Ground Turnaround and Mission Interface and Cargo Handling/Accommodations.

2.0 MISSION ANALYSIS

The mission-analysis effort consisted of (1) establishing the baseline mission (together with nominal excursions from it), (2) collecting and documentating the mission and design requirements and constraints, (3) detailing the mission sequence of events, and (4) assessing alternate mission requirements and their impact on the baseline vehicle design. In addition, the use of an inland launch and recovery site was briefly considered. The results of these studies are reported in this section.

Major mission-analysis tasks, such as cargo handling/passenger accommodations and ground turnaround operations are reported in the special emphasis sections of this report. Launch operations philosophy and major events are detailed in the Launch Operations Plan.

2.1 Mission Profile - The baseline mission is a logistics shuttle mission for the transportation of men and food, tools, equipment, experiments, etc. to a Space Station. The Space Station is parked in a 55-degree-inclined, 270 NM-circular Earth orbit. The major mission events, covering the period from launch to pre-flight readiness for relaunch, are depicted (in simplified form) in Figure 2-1.

The Carrier and the Orbiter are mated together and launched from the Eastern Test Range (ETR). Launch is along a nominal 139-degree azimuth. Staging of the two vehicles occurs at an altitude of about 220,000 feet, 82 nautical miles down-range. Velocity at staging is 14,473 fps (ideal).

After staging, the Carrier rolls, turns, and flies to a landing field located near the launch site. The Carrier is then cycled through the turnaround and recertification operations. The Orbiter, under power, continues its flight into a 45 x 100 NM eliptical orbit. The parking orbit is subsequently circularized to 100 NM. After appropriate phasing, the Orbiter transfers from the parking orbit to the 270 NM Space Station altitude where payload transfer is accomplished.

The Orbiter remains in orbit up to seven days whereupon it returns to Earth loaded with return payload (up to 25,000 lbs.).

At the landing site, the Orbiter is cycled through its recertification phase and is moved to the launch site in preparation for its next flight.

2.2 Mission Constraints and Requirements - A summary of the mission constraints and requirements is given in Table 2-1. The mission requirements are listed by

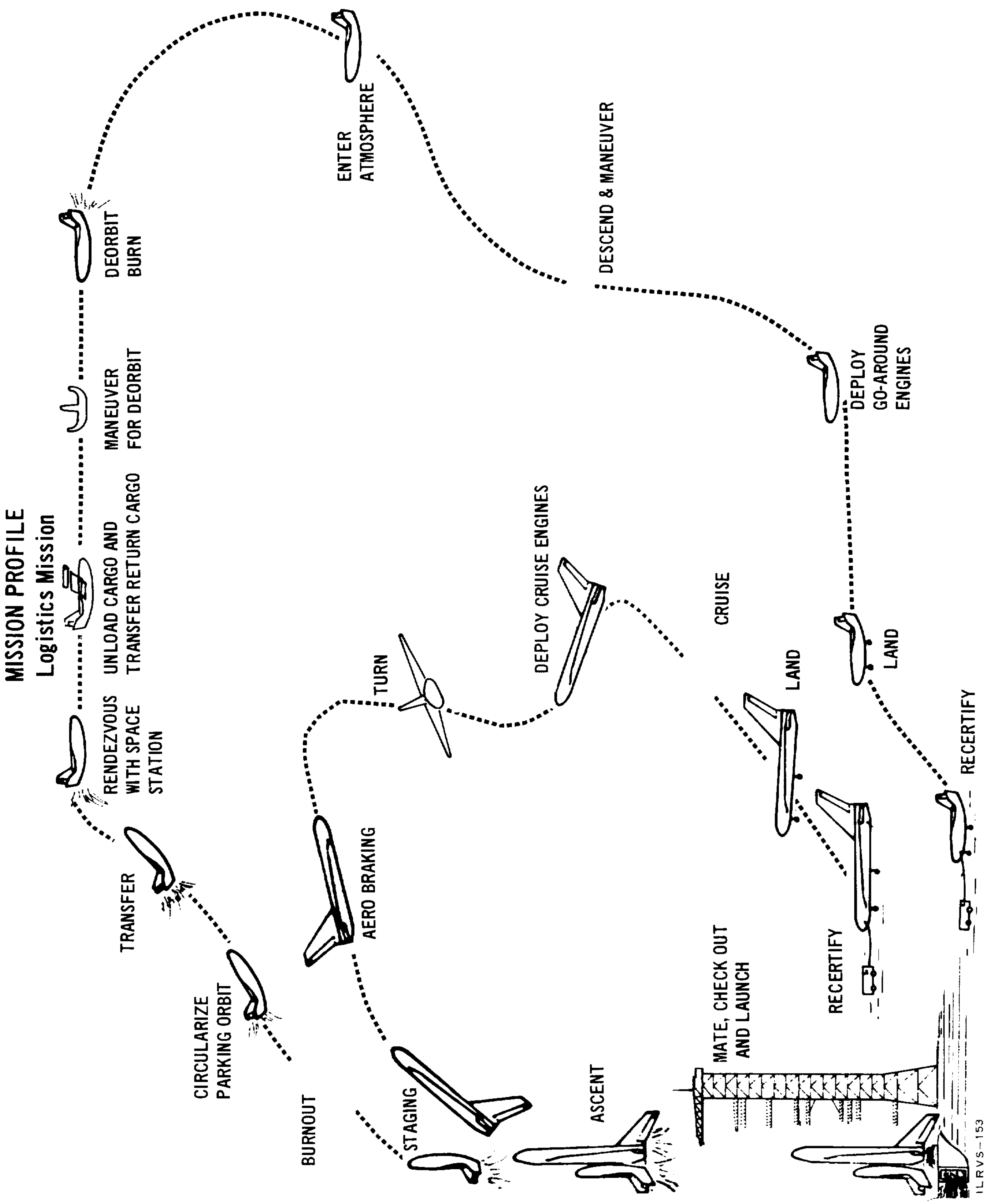

MISSION PROFILE
Logistics Mission
ENTER ATMOSPHERE
DESCEND & MANEUVER
DEORBIT BURN
MANEUVER FOR DEORBIT
UNLOAD CARGO AND TRANSFER RETURN CARGO
RENDEZVOUS WITH SPACE STATION
TRANSFER
CIRCULARIZE PARKING ORBIT
BURNOUT
STAGING
ASCENT
AERO BRAKING
TURN
DEPLOY CRUISE ENGINES
CRUISE
LAND
RECERTIFY
MATE, CHECK OUT AND LAUNCH
DEPLOY GO-AROUND ENGINES
LAND
RECERTIFY
ILRVS-153

Figure 2–1

Table 2–1

MISSION AND DESIGN REQUIREMENTS AND CONSTRAINTS

MISSION EVENT	ITEM	VALUE & UNITS	REMARKS
	TYPE OF MISSION		LOW-EARTH-ORBIT LOGISTICS MISSION
	POSSIBLE ALTERNATE MISSIONS		(1) SATELLITE PLACEMENT/RETRIEVAL (2) DELIVERY OF PROPULSIVE STAGES & PAYLOADS (3) DELIVERY OF PROPELLANTS (4) SATELLITE SERVICE AND MAINTENANCE (5) SHORT-DURATION ORBITAL MISSION
	TIME PERIOD		POST 1974
	TYPE OF VEHICLE		TWO-STAGE FULLY REUSABLE
	BASELINE CANDIDATE VEHICLES (1) FIRST STAGE (CARRIER) (2) SECOND STAGE (ORBITER)		 CLIPPED DELTA HL–10
	CREW SIZE	2 MEN	1 MAN FOR ATMOSPHERIC FLIGHT; 2 MEN FOR ORBITAL OPERATIONS
	NUMBER OF PASSENGERS	10 MEN	PASSENGERS MAY BE OFF-LOADED FOR ADDITIONAL PAYLOAD CAPABILITY
	ORBITAL ALTITUDE (NOM) (RANGE)	 270 N.MI. 200–300 N.MI.	
	ORBITAL INCLINATION (NOM) (RANGE)	 55 DEG 28–90 DEG	
	DISCRETIONARY PAYLOAD (NOM)	 25,000 LB	 INTEGRAL PAYLOAD REMOVABLE IN ORBIT
	PAYLOAD VOLUME	5300 FT^3	CYLINDRICAL CANISTER, 15-FT DIAMETER x 30 FT LENGTH
	ALTERNATE PAYLOADS	10,000 LB 25,000 LB 50,000 LB 50,000 LB	CANISTER, 15-FT DIA, VOLUME – 3000 FT^3 CANISTER, 15-FT DIA x 60-FT LENGTH CANISTER, 15-FT DIA x 60-FT LENGTH CANISTER, 22-FT DIA x 60-FT LENGTH
	RETURN CARGO (MAXIMUM)	25,000 LB	FULL CARGO
	ORBITER OPERATIONAL LIFETIME	100 USES	
	PROBABILITY OF CREW SURVIVAL	0.999	

Table 2–1

MISSION AND DESIGN REQUIREMENTS AND CONSTRAINTS (Continued)

MISSION EVENT	ITEM	VALUE & UNITS	REMARKS
	PROBABILITY OF MISSION COMPLETION	0.950	
	PROBABILITY OF ORBITER SUCCESSFUL RECOVERY	0.990	
	PROBABILITY OF CARRIER SUCCESSFUL RECOVERY	0.995	
	DESIGN FACTOR OF SAFETY	1.4	
	STRUCTURAL TEMPERATURE UNCERTAINTY FACTOR	1.1	
	SINGLE POINT FAILURES		MINIMIZED (OCCUR ONLY IN STRUCTURAL ITEMS)
	AVIONICS		SYSTEM IS PROVIDED WITH ON-BOARD CHECKOUT, AUTONOMOUS FLIGHT CONTROL AND INTEGRAL LAUNCH/ON-ORBIT/REENTRY ELECTRONICS SUBSYSTEMS
	ORBITER INTERNAL ENVIRONMENT		"SHIRTSLEEVE" CONDITIONS, i.e., NO SUITS
0.0 PRELAUNCH OPERATIONS			
0.1 PRELAUNCH OPERATIONS	NUMBER OF LAUNCH SITES	1	
	PRELAUNCH OPERATIONS TIME	24 HR	3-SHIFT OPERATIONS; PHASED WITH RECYCLING OPERATIONS
	LAUNCH RATE(S)	4, 8, 12 LAUNCHES/YR 10,30,50,100 LAUNCHES/YR	MINIMUM OF 4 FLIGHTS/YR; 8 & 12 ARE NOMINAL VALUES; MAXIMUM VALUE TO BE DETERMINED BY ECONOMIC CONSIDERATIONS
	LAUNCH PAD RESPONSE TIME	RAPID	TO APPROACH 24 HR
1.0 ASCENT PHASE			
1.1 LAUNCH	LAUNCH AZIMUTH	44–110 DEG	ETR UNMODIFIED
	CONSTRAINTS	35–180 DEG	ETR MODIFIED
	MAXIMUM BOOST DYNAMIC PRESSURE	450 PSF	
	WINDS		95-PERCENTILE WINDS FOR LOADS ANALYSES
	MAX α –q DURING LAUNCH	6000 PSF DEG	
	MAXIMUM BOOST ACCELERATION	3 g's 4 g's	WITH PASSENGERS WITH PAYLOAD
	TOTAL LAUNCH-INTO-ORBIT ΔV	31,250 FPS	IDEAL VELOCITY
	LAUNCH ABORT		INTACT – EXCEPT FOR INITIAL 20 SEC FOLLOWING LIFTOFF
	SEPARATE LAUNCH ESCAPE SYSTEMS		REQUIRED ONLY FOR DEVELOPMENT FLIGHTS

Table 2-1
MISSION AND DESIGN REQUIREMENTS AND CONSTRAINTS (Continued)

MISSION EVENT	ITEM	VALUE & UNITS	REMARKS
1.0 ASCENT PHASE (CONTINUED)			
	ENGINE-OUT CAPABILITY		BOTH STAGES
	ON-PAD ABORT MODES		CARRIER: ESCAPE HATCH ORBITER: ESCAPE HATCH
	ABORT PRIORITIES		(1) CREW SAFETY (2) SPACECRAFT REUSABILITY (ONLY AFTER (1) IS ASSURED)
1.2 STAGING	1ST STAGE BURN TIME	206 SEC	ΔV_1 = 14,420 FPS (IDEAL)
	2ND STAGE BURN TIME	224 SEC	ΔV_2 = 16,777 FPS (IDEAL)
1.3 PARKING ORBIT INSERTION & COAST	PARKING ORBIT PARAMETERS	45 x 100 N.MI.	LOW-ENERGY, LOW-TEMPERATURE ASCENT
1.4 PARKING ORBIT CIRCULARIZATION	FINAL PARKING ORBIT ALTITUDE	100 N.MI.	
	CIRCULARIZATION ΔV	100 FPS	TOTAL ON-ORBIT ΔV = 2000 FPS; INCLUDES CONTINGENCIES. (SEE TABLE 2-2 FOR FURTHER BREAKDOWN)
1.5 PARKING ORBIT COAST	COAST TIME	0–20 HR	ORBITER PHASES WITH SPACE STATION
1.6 GROSS RENDEZVOUS	TRANSFER ΔV REQUIREMENTS	600–820 FPS	WITH 1^0 PLANE CHANGE CAPABILITY; TRANSFER TO 255 N.MI. ORBIT; 2- IMPULSE MANEUVER; SPACECRAFT APPROX 15 N.MI. BELOW AND 75 N.MI. BEHIND STATION
	TRANSFER TIME	45–95 MIN	
	TYPE OF TRANSFER		MODIFIED LIMITED RENDEZVOUS
	TOTAL ASCENT TIME ALLOWABLE	24 HR	MAXIMUM
1.7 TERMINAL RENDEZVOUS	TRANSFER ΔV REQUIREMENT	200 FPS	2-IMPULSE MANEUVER; SPACECRAFT APPROX 3 N.MI.
	TRANSFER TIME	24–66 MIN	BEHIND STATION
2.0 ORBITAL OPERATIONS PHASE			
2.1 UNLOAD PAYLOAD	METHOD		NO EVA; "SHIRTSLEEVE" OPERATION; SEE CARGO HANDLING/ PASSENGER ACCOMMODATIONS SPECIAL EMPHASIS SECTION
2.2 DOCK PAYLOAD TO SPACE STATION	ΔV REQUIRED	40 FPS	
	METHOD		USE OF SPACE TUG; SEE CARGO HANDLING SPECIAL EMPHASIS SECTION

Table 2-1
MISSION AND DESIGN REQUIREMENTS AND CONSTRAINTS (Continued)

MISSION EVENT	ITEM	VALUE & UNITS	REMARKS
2.3 PASSENGER TRANSFER	METHOD		INTACT WITH PAYLOAD
2.4 CREW TRANSFER	METHOD		INTACT WITH PAYLOAD; NOT FIRM BASELINE
2.5 PERFORM SPACE STATION SUPPORT OPERATIONS	SPACE STATION SUPPORT OPERATIONS		STATION KEEPING; STATION MAINTENANCE; DELIVERY & COLLECTION OF REMOTE HARDWARE, ETC.
3.0 DESCENT PHASE			
3.1 DEORBIT BURN	ΔV REQUIRED	500 FPS	
3.2 ATMOSPHERIC ENTRY	ENTRY ALTITUDE	400,000 FT	
	ENTRY VELOCITY	25,990 FPS	
	ENTRY ANGLE	-1.5 DEG	
3.3 DEPLOY GO-AROUND ENGINES	ALTITUDE	30,000 FT	
3.4 LANDING	TIME FROM ENTRY	~1 HR	
	LANDING SITE(S)		
	CARRIER		IN THE VICINITY OF ETR (PRIMARY)
	ORBITER		IN THE VICINITY OF ETR
	LANDING CONSTRAINTS		ALL WEATHER, DAY OR NIGHT, POWER-ON, ONCE-AROUND (ORBITER). POWER- OFF EMERGENCY LANDING CAPABILITIES
	RETURN FREQUENCY CAPABILITY		AT LEAST ONCE EVERY 24-HR PERIOD
	TOTAL DESCENT TIME ALLOWABLE	24 HR	
	CREW RECOVERY TIME	24 HR	MAXIMUM TIME AFTER TOUCHDOWN; NOMINAL TIME IS AS SOON AS POSSIBLE; RECOVERY AIDS AND SURVIVAL GEAR PROVIDED FOR CREW FOR A 24-HR PERIOD
	EMERGENCY RECOVERY		WATER RECOVERY CAPABILITY
	SPACECRAFT RECOVERY		AS SOON AS FEASIBLE; CREW RECOVERY HAS FIRST PRIORITY
4.0 MAINTENANCE OPERATIONS			
4.1 PERFORM IMMEDIATE POST-FLIGHT MAINTENANCE	TIME REQUIRED	27.5 HR	3-SHIFT OPERATION*
4.2 PERFORM PRE-FLIGHT MAINTENANCE	TIME REQUIRED	120 HR	2-SHIFT OPERATION*
4.3 PERFORM PRE-LAUNCH OPERATIONS	TIME REQUIRED	24 HR	3-SHIFT OPERATION*

*TIMES REPRESENT 20-TH OPERATION @ 90% LEARNING.

mission phase and major event for ease in back referencing. The requirements and constraints were drawn from both the Program Study Outline (PSO) and from MDAC investigations during the study.

Major items among the mission requirements include the mission-altitude range of 200 to 300 NM (nominally, 270 NM) and the mission orbital inclination range of 28 to 90 degrees (nominally, 55 degrees). The nominal mission payload consists of a cylindrically shaped canister having a diameter of 15 feet and a length of 30 feet. Alternate payloads are identified in the chart. Possible alternate missions are also itemized here.

The impulsive-velocity requirements are listed for each phase in which they occur, as determined during the study. As a reference, the NASA supplied on-orbit impulsive-velocity requirements are shown separately in Table 2-2. A total of 2000 fps is allowed for orbital maneuvering, including parking orbit circularization, orbit-to-orbit transfer, and the deorbit burn. Greater levels of detail can be found in the appropriate sections of this report.

2.3 <u>Logistics Requirements</u> - Annual logistics requirements are shown in Table 2-3 for 6, 9, 12, 18, and 24-man space stations. The requirements for the 6, 9, 18, and 24-man stations were established during the MDAC Advanced Logistic System Study (ALSS), which was performed for the NASA (October 1967). The 12-man station data, which is of prime interest in this study, was interpolated from the ALSS data.

Annual experiment equipment and supply requirements are detailed in Table 2-4 for typical 6, 9, 12, 18, and 24-man space stations. The equipment is grouped into six general categories, namely,

- o Astronomy
- o Earth resources
- o Meteorology
- o Biology
- o Long-term flight
- o Advanced systems and equipment technology

For the 12-man space station, the total annual equipment requirements were determined to weigh about 56,350 lbs.

Table 2-2

ON-ORBIT IMPULSIVE VELOCITY REQUIREMENTS[1]

EVENT	VELOCITY REQUIRED (fps)
1. Circularize at 100 NM[2]	100
2. Transfer into 260 NM Phasing Orbit[3]	558
3. Terminal Rendezvous & Docking	142
4. Launch Dispersion and Plane Change	200
5. Deorbit	500
6. Contingencies	500
TOTAL	2000 fps

[1] NASA - provided

[2] After insertion into 45 x 100 NM orbit

[3] Rendezvous within 24 hours.

Table 2-3

SPACE STATION ANNUAL LOGISTICS REQUIREMENTS

ITEM		REQUIREMENT (LBS) SPACE STATION CREW SIZE 6 MEN	9 MEN	12 MEN	18 MEN	24 MEN
HUMAN EXPENDABLES						
o PERSONAL SUPPLIES		1,040	1,570	2,080	3,140	4,160
o METABOLIC OXYGEN		4,380	6,570	8,760	13,120	16,000
o LiOH (BACK PACK)		420	630	840	1,200	1,600
o LIFE SUPPORT SYSTEM		200	420	540	840	1,120
o FOOD		4,000	7,200	9,200	14,400	19,200
o WATER		900	1,300	1,700	2,400	3,200
	SUBTOTAL	11,820	17,690	23,120	34,100	45,280
STATION EXPENDABLES						
o OXYGEN LEAKAGE		1,600	1,600	1,600	1,600	1,600
o NITROGEN LEAKAGE		5,000	5,000	5,000	5,000	5,000
o RCS PROPELLANT		7,610	7,610	7,610	8,750	8,750
	SUBTOTAL	14,210	14,210	14,210	15,350	15,350
TANKAGE						
o OXYGEN		760	1,050	1,260	1,750	2,300
o NITROGEN		2,750	2,750	2,750	2,750	2,750
o PROPELLANT		5,180	5,180	5,180	5,400	5,400
o WATER		80	120	160	250	320
	SUBTOTAL	8,770	9,100	9,350	10,150	10,770
SPARES		3,000	3,000	3,000	3,600	3,600
TOTAL LESS EXPERIMENT EQUIPMENT		37,800	44,000	49,680	63,200	75,000
EXPERIMENT EQUIPMENT & SUPPLIES		45,000	52,000	56,360	66,100	77,700
	TOTAL	82,800	96,000	106,030	129,300	152,700

Table 2-4

SPACE STATION ANNUAL EXPERIMENT EQUIPMENT AND SUPPLY REQUIREMENTS

EQUIPMENT CATEGORY	ITEM	REQUIREMENT (LBS) SPACE STATION CREW SIZE 6 MEN	9 MEN	12 MEN	18 MEN	24 MEN
1. ASTRONOMY	o FILM AND TAPE	100	100	100	200	200
	o SPARES	3,400	3,900	4,200	4,800	5,800
	o FUEL-CABIN AIR	500	600	750	800	900
	SUBTOTAL	4,000	4,600	5,050	5,800	6,900
2. EARTH RESOURCES	o FILM	13,900	16,000	17,300	20,400	24,000
	o SPARES	3,450	4,000	4,400	5,100	6,000
	SUBTOTAL	17,350	20,000	21,700	25,500	30,000
3. METEOROLOGY	o FILM AND TAPE	1,000	1,200	1,300	1,500	1,800
	o SPARES	300	300	300	400	450
	SUBTOTAL	1,300	1,500	1,600	1,900	2,250
4. BIOLOGY	o INSTRUMENTS, TEST EQUIPMENT AND BIOLOGICAL SPECIMENS	5,200	6,000	6,500	7,600	9,000
5. LONG TERM FLIGHT	o BIOMEDICAL EQUIPMENT & SUPPLIES	100	100	100	100	100
	o LIFE SUPPORT SYSTEM EQUIP.	450	500	520	560	750
	o POWER EQUIPMENT	550	600	650	800	900
	o CONTROL EQUIPMENT	900	1,060		1,400	1,600
	o CREW SYS EQUIP & SUPPLIES	8,200	9,400	11,500	12,100	14,100
	SUBTOTAL	10,200	11,660	12,770	14,960	17,500
6. ADVANCE TECHNOLOGY OF SYSTEMS AND EQUIPMENT	o STRUCTURES	2,500	2,840	3,100	3,600	4,260
	o STABILIZATION & CONTROL	2,700	3,110	3,350	4,000	4,660
	o FLUID SYSTEMS	400	500	550	650	750
	o MATERIALS & COATINGS	280	320	350	420	480
	o SUBSYSTEM DEVELOPMENT	750	860	930	1,100	1,290
	o ENVIRONMENTAL MEASUREMENT	400	420	450	530	630
	SUBTOTAL	7,030	8,050	8,730	10,300	12,070
	TOTAL	45,080	51,810	56,350	66,060	77,720

The factors used in deriving both human and space station logistics requirements are presented in Table 2-5. Human expendable factors consist of personal supplies, metabolic oxygen, lithium hydroxide, the life support system, food, and make-up water. Space station expendables considered were propellants and oxygen and nitrogen requirements due to leakage.

Note that, from Table 2-3, the total annual 12-man space station logistic requirement is given as 106,000 pounds. This figure is compatible with the study nominal payload weight of 25,000 lbs for a launch rate of four launches per year.

2.4 Mission Sequence of Events - A detailed timeline of major mission events for the baseline logistics mission is given in Table 2-6. The mission events are grouped according to the following mission phases:

- Prelaunch Operations
- Ascent
- Orbital Operations
- Orbiter Descent
- Carrier Descent
- Maintenance Operations

Both event-initiation times and event-duration times are presented in this table. Note that the tiems shown in this chart sometimes overlap, indicating the parallel occurrance of events.

Discussions as to the philosophical approaches used in determining the exhibited times are found in the appropriate sections of this report. It should also be noted that the nominal values of the mission-event times are shown in Table 2-6. The minimum prelaunch and maintenance event times, for example, are discussed in the ground turnaround analyses of Section 4.1. For purposes of programmatic analyses, the nominal times of Table 2-4 were used. However, for the baseline program requirements, total program costs are found to be relatively insensitive to ground turnaround duration.

2.5 Alternate Mission Capability - The capability of the baseline system to perform the reference and alternate missions was investigated. A procedure for assessing a space vehicle's mission capability was developed and is outlined in the following paragraphs.

Table 2-5

SPACE STATION SUPPORT REQUIREMENTS

A. FACTORS USED IN DERIVING HUMAN EXPENDABLES

ITEM	REQUIREMENTS	REMARKS
PERSONAL SUPPLIES	0.475 LB/MAN/DAY	PERSONAL GEAR AND MEDICAL SUPPLIES ONLY. INCLUDES PACKAGING.
METABOLIC OXYGEN	2.0 LB/MAN/DAY	PACKAGING WEIGHT, INCLUDED IN TANKAGE. NONREGENERATIVE SYSTEM REQUIREMENT.
LiOH	0.19 LB/MAN/DAY	PACKAGING WEIGHT: INCLUDED IN TANKAGE.
LIFE SUPPORT SYSTEM	0.13 LB/MAN/DAY	PACKAGING WEIGHT: INCLUDED IN TANKAGE.
FOOD	2.2 LB/MAN/DAY	INCLUDES PACKAGING
WATER (MAKE-UP)	0.4 LB/MAN/DAY	PACKAGING WEIGHT: INCLUDED IN TANKAGE. MAKE-UP WATER ONLY.

B. FACTORS USED IN DERIVING STATION EXPENDABLES

ITEM	REQUIREMENTS	REMARKS
OXYGEN LEAKAGE	4.2 LB/DAY	PACKAGING WEIGHT: INCLUDED IN TANKAGE.
NITROGEN LEAKAGE	13.8 LB/DAY	PACKAGING WEIGHT: INCLUDED IN TANKAGE.
PROPELLANT	21 LB/DAY (6-12 MAN STATION) 24 LB/DAY (18-24 MAN STATION)	PACKAGING WEIGHT: INCLUDED IN TANKAGE.

Table 2-6

SEQUENCE OF EVENTS
TWO-STAGE FULLY REUSABLE CONCEPT

MISSION EVENT	EVENT INITIATION TIME	EVENT DURATION	REMARKS
0.0 PRELAUNCH OPERATIONS PHASE			
0.1 Transport Carrier to Pad	T_o - 24 hrs.	1 hr.	
0.2 Erect Carrier on Pad	T_o - 23 hrs.	5 hrs.	
0.3 Transport Orbiter to Pad	T_o - 18 hrs.	1 hr.	
0.4 Erect Orbiter on Pad	T_o - 17 hrs.	4 hrs.	
0.5 Mate Orbiter to Carrier	T_o - 13 hrs.	3 hrs.	
0.6 Connect Holddowns	T_o - 10.5 hrs.	30 min.	
0.7 Connect Cryogenic Service Lines	T_o - 10 hrs.	2 hrs.	
0.8 Perform Tank Leakage Test	T_o - 8 hrs.	2 hrs.	
0.9 Power Up for Range Check and Navigational Input	T_o - 6 hrs.	2 hrs.	
0.10 Propulsion Subsystem Operational Checkout	T_o - 6 hrs.	2 hrs.	
0.11 Final Preparation and Inspection	T_o - 4 hrs.	1 hr.	
0.12 Crew Ingress	T_o - 3.5 hrs.	20 min.	Both on Carrier & Orbiter.
0.13 Final Systems Checkout and Guidance Update	T_o - 3 hrs.	1 hr.	
0.14 Crew Egress	T_o - 2 hrs.	15 min.	Safety measure

Table 2-6

SEQUENCE OF EVENTS
TWO-STAGE FULLY REUSABLE CONCEPT (CONTINUED)

MISSION EVENT	EVENT INITIATION TIME	EVENT DURATION	REMARKS
0.15 Begin Cryogenic Servicing	T_o - 2 hrs.	78 min.	
0.16 Begin LH_2 Precool	T_o - 120 min.	15 min.	
0.17 Begin LH_2 Slow Fill	T_o - 105 min.	12 min.	To 4%.
0.18 Begin LH_2 Fast Fill	T_o - 93 min.	32 min.	To 95%.
0.19 Begin LO_2 Precool	T_o - 87 min.	15 min.	
0.20 Begin LO_2 Slow Fill	T_o - 72 min.	6 min.	To 5%.
0.21 Begin LO_2 Fast Fill	T_o - 66 min.	20 min.	To 96.5%.
0.22 Begin LH_2 Slow Fill (Topping)	T_o - 61 min.	15 min.	To 100%.
0.23 Begin LO_2 Slow Fill (Topping)	T_o - 46 min.	4 min.	To 100%.
0.24 Crew Ingress	T_o - 40 min.	20 min.	
0.25 Passenger Ingress	T_o - 35 min.	20 min.	
0.26 GSE Removal	T_o - 15 min.	5 min.	
0.27 Final Countdown	T_o - 10 min.	10 min.	

Table 2-6
SEQUENCE OF EVENTS
TWO-STAGE FULLY REUSABLE CONCEPT (CONTINUED)

MISSION EVENT	EVENT INITIATION TIME	EVENT DURATION	REMARKS
1.0 ASCENT PHASE			
1.1 Ignite Engines	T_o - 3 sec.	-	
1.2 Holddown Space Vehicle	T_o - 3 sec.	3 sec.	
1.3 Release Holddowns (Lift-off)	T_o	-	
1.4 Translate Engine Nozzles	T_o + 54 sec.	-	
1.5 Experience Max.-q	T_o + 75 sec.	-	Max.-q = 420 psf.; $h \cong$ 32,000 ft.
1.6 Begin Constant 3g Acceleration	T_o + 167 sec.	39 sec.	$r \cong$ 35 NM; $h \cong$ 160,000 ft.
1.7 Separate Stages	T_s	-	$T_s = T_o + 206$ sec.; $h \cong$ 220,000 ft.; $r \cong$ 82 NM; ΔV_1 = 14,473 fps(ideal);
1.8 Burn Orbiter into Parking Orbit	T_s	224 sec.	ΔV_2 = 16,777 fps(ideal); $r \cong$ 660 NM; 45 x 100 NM parking orbit
1.9 Begin Carrier Descent Phase	T_s	-	
1.10 Maintain Constant 3g Orbiter Acceleration	T_s + 168 sec.	56 sec.	
1.11 Insert Orbiter into Parking Orbit	T_s + 224 sec.	-	
1.12 Circularize Parking Orbit	T_s + 45.4 min.	-	Circularize to 100 NM ΔV = 100 fps.
1.13 Coast in Parking Orbit	T_s + 45.4 min.	0-20 hrs.	Phasing with Space Station.

Table 2-6
SEQUENCE OF EVENTS
TWO-STAGE FULLY REUSABLE CONCEPT (CONTINUED)

MISSION EVENT	EVENT INITIATION TIME	EVENT DURATION	REMARKS
1.14 Transfer to 255-NM orbit	T_s + 21 hrs.	96 min. 2	3-Impulse maneuver; ΔV = 590 fps; S/C approx. 15 NM below and 75 NM behind station.
1.15 Coast in 255-NM orbit	T_s + 22.5 hrs.	30 min.	Phasing with space station.
1.16 Transfer to 270-NM orbit	T_s + 23 hrs.	66 min.	2-Impulse maneuver; ΔV = 200 fps; S/C approx. 3 NM behind station.

Table 2-6

SEQUENCE OF EVENTS
TWO-STAGE FULLY REUSABLE CONCEPT (CONTINUED)

MISSION EVENT	EVENT INITIATION TIME	EVENT DURATION	REMARKS
2.0 ORBITAL OPERATIONS PHASE			
2.1 Checkout Orbiter Subsystems	T_x	30 min	Quick check using OCS.
2.2 Pressurize Crew-Access Tunnel	T_x + 30 min.	1 hr.	
2.3 Switch Orbiter Subsystems to Standby	T_x + 90 min.	30 min.	
2.4 Crew Transfers to Payload Canister	T_x + 2 hrs.	2 hrs.	
2.5 Open Cargo Bay Doors	T_x + 4 hrs.	12 min.	
2.6 Translate and Hold Payload Canister Outward	T_x + 4.2 hrs.	24 min.	
2.7 Dock Space Tug with Payload	T_x + 4.6 hrs.	24 min.	
2.8 Release Payload	T_x + 5 hrs.	-	
2.9 Retract Translational Devices	T_x + 5 hrs.	12 min.	
2.10 Close Cargo Bay Doors	T_x + 5.2 hrs.	12 min.	
2.11 Rendezvous with Space Station	T_x + 5.6 hrs.	1 hr.	$\Delta V \doteq 10$ fps.
2.12 Dock Payload to Space Station	T_x + 6.6 hrs.	24 min.	$\Delta V \doteq 30$ fps
2.13 Undock Space Tug	T_x + 7 hrs.	-	
2.14 Dock Space Tug to Space Station	T_x + 7 hrs.	30 min.	

Table 2-6
SEQUENCE OF EVENTS
TWO-STAGE FULLY REUSABLE CONCEPT (CONTINUED)

MISSION EVENT		EVENT INITIATION TIME	EVENT DURATION	REMARKS
2.15	Perform On-Orbit Operational Support to Space Station	T_x + 7.5 hrs.	5 days	
2.16	Undock Space Tug from Space Station	T_y	-	$T_y \cong T_x + 5$ days
2.17	Dock Space Tug to Return Payload	T_y + 30 min.	30 min.	
2.18	Undock Payload from Space Station	T_y + 1 hr.	-	
2.19	Rendezvous with Orbiter	T_y + 1 hr.	1 hr.	
2.20	Open Cargo Bay Doors	T_y + 2 hrs.	12 min.	
2.21	Extend Payload Translational Devices	T_y + 2.2 hrs.	12 min.	
2.22	Lock Holding Arms onto Payload	T_y + 2.4 hrs.	12 min.	
2.23	Undock Space Tug from Payload	T_y + 2.6 hrs.	-	
2.24	Withdraw Payload into Orbiter	T_y + 2.6 hrs.	12 min.	
2.25	Close Cargo Bay Doors	T_y + 2.8 hrs.	12 min.	
2.26	Pressurized Crew-Access Tunnel	T_y + 3 hrs.	1 hr.	
2.27	Crew Transfers to Crew Cabin	T_y + 4 hrs.	2 hrs.	
2.28	Switch Orbiter Subsystems to Fully Active Status	T_y + 6 hrs.	1 hr.	
2.29	Checkout Orbiter Subsystems	T_y + 7 hrs.	2 hrs.	Quick check using OCS.

Table 2-6
SEQUENCE OF EVENTS
TWO-STAGE FULLY REUSABLE CONCEPT (CONTINUED)

MISSION EVENT	EVENT INITIATION TIME	EVENT DURATION	REMARKS
4.0 MAINTENANCE PHASE			
4.1 Crew Egress	T_L + 10 min.	20 min.	
4.2 Passenger Egress*	T_L + 10 min.	20 min.	
4.3 Data Removal	T_L + 30 min.	10 min.	Critical Data Only.
4.4 Install Safety Devices	T_L + 30 min.	12 min.	All propulsive and pyrotechnic systems safed.
4.5 Make Visual Inspection	T_L + 42 min.	15 min.	
4.6 Perform Cabin Switch Check	T_L + 48 min.	15 min.	
4.7 Cool and Decontaminate S/C	T_L + 63 min.	3 hrs.	
4.8 Move S/C to post-flight Maintenance Area	T_L + 4 hrs.	1 hr.	
4.9 Position Emergency Equipment	T_L + 5 hrs.	15 min.	Precautionary measure.
4.10 Provide Acces to Required Areas	T_L + 5 hrs.	30 min.	
4.11 Remove Cargo*	T_L + 5.5 hrs.	1.5 hrs.	
4.12 Deservice ACS	T_L + 7 hrs.	7 hrs.	
4.13 Deservice Propulsion System	T_L + 14 hrs.	12.5 hrs.	
4.14 Deservice APU System	T_L + 14 hrs.	7 hrs.	
4.15 Deservice Fuel Cells	T_L + 14 hrs.	7 hrs.	
4.16 Move S/C to Pre-Flight Maintenance Area	T_L + 26.5 hrs.	1 hr.	

* Pertains to 2nd stage (Orbiter) only.

Table 2-6

SEQUENCE OF EVENTS
TWO-STAGE FULLY REUSABLE CONCEPT (CONTINUED)

MISSION EVENT	EVENT INITIATION TIME	EVENT DURATION	REMARKS
3.0B CARRIER DESCENT PHASE			
3.1B Shutdown Boost Engines	T_s	-	$h \cong 220,000$ ft. $r \cong 82$ NM
3.2B Begin 180-degree Roll	$T_s + 10$ sec.	90 sec.	
3.3B Begin Aerodynamic Braking	$T_s + 100$ sec.	-	
3.4B Begin Unpowered Return	$T_L - 5860$ sec.	5860 sec.	$h \cong 200,000$ ft. $r \cong 420$ NM
3.5B Deploy Cruise Engines	$T_L - 600$ sec.	-	
3.6B Decelerate to 175 KIAS	$T_L - 354$ sec.	-	$h \cong 20,000$ ft. $r \cong 20$ NM
3.7B Start Cruise Engines	$T_L - 336$ sec.	-	$r \cong 19$ NM
3.8B Burn Cruise Engines	$T_L - 336$ sec.	336 sec.	
3.9B Level Off	$T_L - 162$ sec.	-	$h \cong 2,000$ ft. $r \cong 8$ NM
3.10B Extend Landing Gear	$T_L - 162$ sec.	-	
3.11B Intercept 3-degree Glide Slope	$T_L - 144$ sec.	-	
3.12B Landing	T_L	-	$T_L \cong T_s + 2$ hrs.

Table 2-6

SEQUENCE OF EVENTS
TWO-STAGE FULLY REUSABLE CONCEPT (CONTINUED)

MISSION EVENT	EVENT INITIATION TIME	EVENT DURATION	REMARKS
3.0A ORBITER DESCENT PHASE			
3.1A Maneuver for Deorbit	T_y + 9 hrs.	30 min.	
3.2A Fire Deorbit Motor	T_D	-	ΔV = 425 fps; $T_D = T_y + 9.5$ hrs.
3.3A Burn Deorbit Motor	T_D	10 sec.	Maximum time.
3.4A Terminate Deorbit Burn	T_D + 10 sec.	-	
3.5A Descent from Orbit	T_D + 10 sec.	32 min.	Descend to 400,000 ft., Entry Angle = -1.5 degrees.
3.6A Maneuver for Entry	T_D + 2 min.	30 min.	Maximum time.
3.7A Enter Atmosphere	T_E	-	$T_E = T_D + 32$ min.
3.8A Descend Through Atmosphere	T_E	3670 sec.	
3.9A Deploy Go-Around Engines	T_E + 3510 sec.	-	h = 30,000 ft.
3.10A Start Go-Around Engines	T_E + 3520 sec.	-	
3.11A Burn Go-Around Engines	T_E + 3520 sec.	150 sec.	
3.12A Extend Landing Gear	T_E + 3590 sec.	-	h = 1000 ft.
3.13A Land	T_L	-	$T_L = T_E + 3670$ sec.

Table 2-6

SEQUENCE OF EVENTS
TWO-STAGE FULLY REUSALBE CONCEPT (CONTINUED)

MISSION EVENT	EVENT INITIATION TIME	EVENT DURATION	REMARKS
4.17 Install S/C on Handling/Transporting Vehicle	T_L + 27.5 hrs.	1 hr.	
4.18 Position Maintenance AGE	T_M	30 min.	$T_M = T_L + 28.5$ hrs.
4.19 Begin Quality Assurance Inspection	T_M + 30 min.	119.5 hrs.	Continuous.
4.20 Checkout S/C Subsystems	T_M + 30 min.	4.5 hrs.	Maximum use is made of onboard checkout system. (OCS).
4.21 Provide Access to Required Areas	T_M + 5 hrs.	8 hrs.	
4.22 Perform Scheduled Maintenance	T_M + 13 hrs.	80 hrs.	
4.23 Perform Unscheduled Maintenance	T_M + 13 hrs.	80 hrs.	Maximum time.
4.24 Close-up Access Areas	T_M + 93 hrs.	3 hrs.	
4.25 Perform Post-Maintenance Procedures	T_M + 96 hrs.	24 hrs.	
4.26 Load S/C on Erector/Transporter	T_M + 113 hrs.	1.5 hrs.	
4.27 Load Crew Provisions on S/C	T_M + 114.5 hrs.	30 min.	
4.28 Load Payload Canister on S/C*	T_M + 115 hrs.	3 hrs.	Less passengers.
4.29 Load Jet Fuel	T_M + 118 hrs.	2 hrs.	

Begin PRELAUNCH OPERATIONS PHASE (See Phase 0.0)

* Pertains to 2nd stage (Orbiter) only.

2.5.1 Payload Sensitivity to On-Orbit Impulsive Velocity Requirements - Payload weight sensitivity to on-orbit impulsive velocity requirements, is shown in Figure 2-2 for the baseline system. Here, payload is traded pound for pound with decreased propellant requirements when operating on the left side of the design point. On the right side, an increase of one pound of propellant results in a decrease of 1.15 pounds of cargo. The 1.15 factor accounts for both propellant and added inerts, where propellant tankage is assumed to be installed in the payload canister.

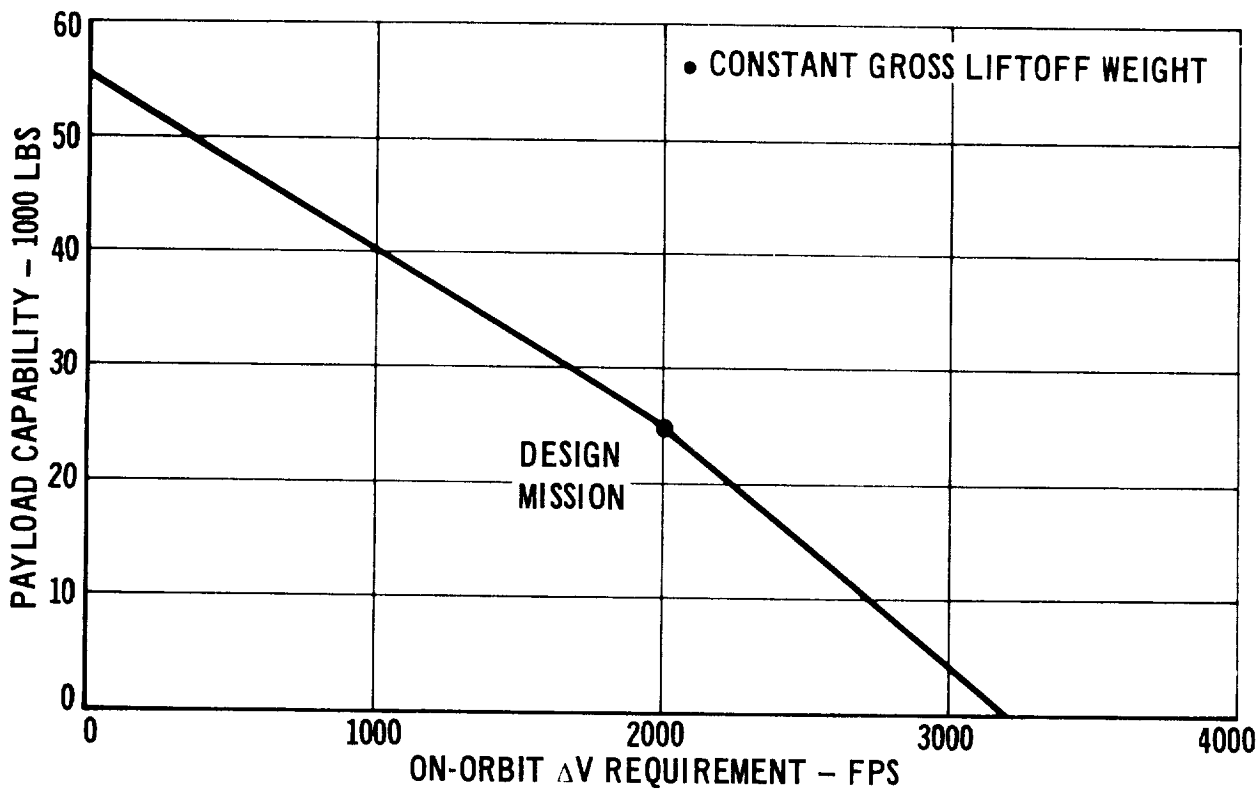
Since the baseline vehicle was designed to return a maximum 25,000 pounds of payload, any additional return payload would have to be left in orbit. Thus, it is conceivable, depending on on-orbit impulsive velocity requirements, that the system could launch over 50,000 lbs into some particular orbits.

Referring to Figure 2-2, in the area of the design point, sensitivities of approximately 17.8 lbs of cargo per 1 fps of impulsive velocity requirement are seen on the left, and 19.4 lbs/fps on the right. The design point reflects an impulsive velocity requirement of 2000 fps for on-orbit maneuvering. It should be noted that if the cargo quantity transported into orbit is also returned and corresponding orbiter subsystem modifications taken into account, a 15 lb/fps exchange would be approximately true for either side at the design point in Figure 2-2. Note also that the chart assumes a constant gross liftoff weight.

2.5.2 Mission Performance Capability - The baseline vehicle was designed to deliver 25,000 lbs at payload into a 270 NM orbit inclined at 55 degrees via a 100 NM circular parking orbit. The additional velocity increment (or decrement) required to attain other inclinations is shown in Figure 2-3. If Hohmann transfer to mission altitudes from a 100 NM circular orbit is assumed, Figure 2-4 may be used to determine the additional impulsive-velocity requirements. Thus, Figures 2-2, 2-3, and 2-4 can be used to determine the system's mission-performance capability, where payload is traded for propellant (or impulsive-velocity capability).

The procedure is as follows. First, impulsive-velocity requirements for a number of delivered payloads over the range of interest are read from Figure 2-2. Next, a value of 1410 fps (2000 - 590) is subtracted from each of these impulsive-velocity requirements. The 1410 fps accounts for all on-orbit impulsive-velocity requirements other than transfer (gross rendezvous). In this manner, the baseline case is zeroed and all other cases are scaled accordingly. Thus, the 25,000 lb.

PAYLOAD SENSITIVITY TO ON-ORBIT Δ V REQUIREMENTS



ILRVS–232F

Figure 2–2

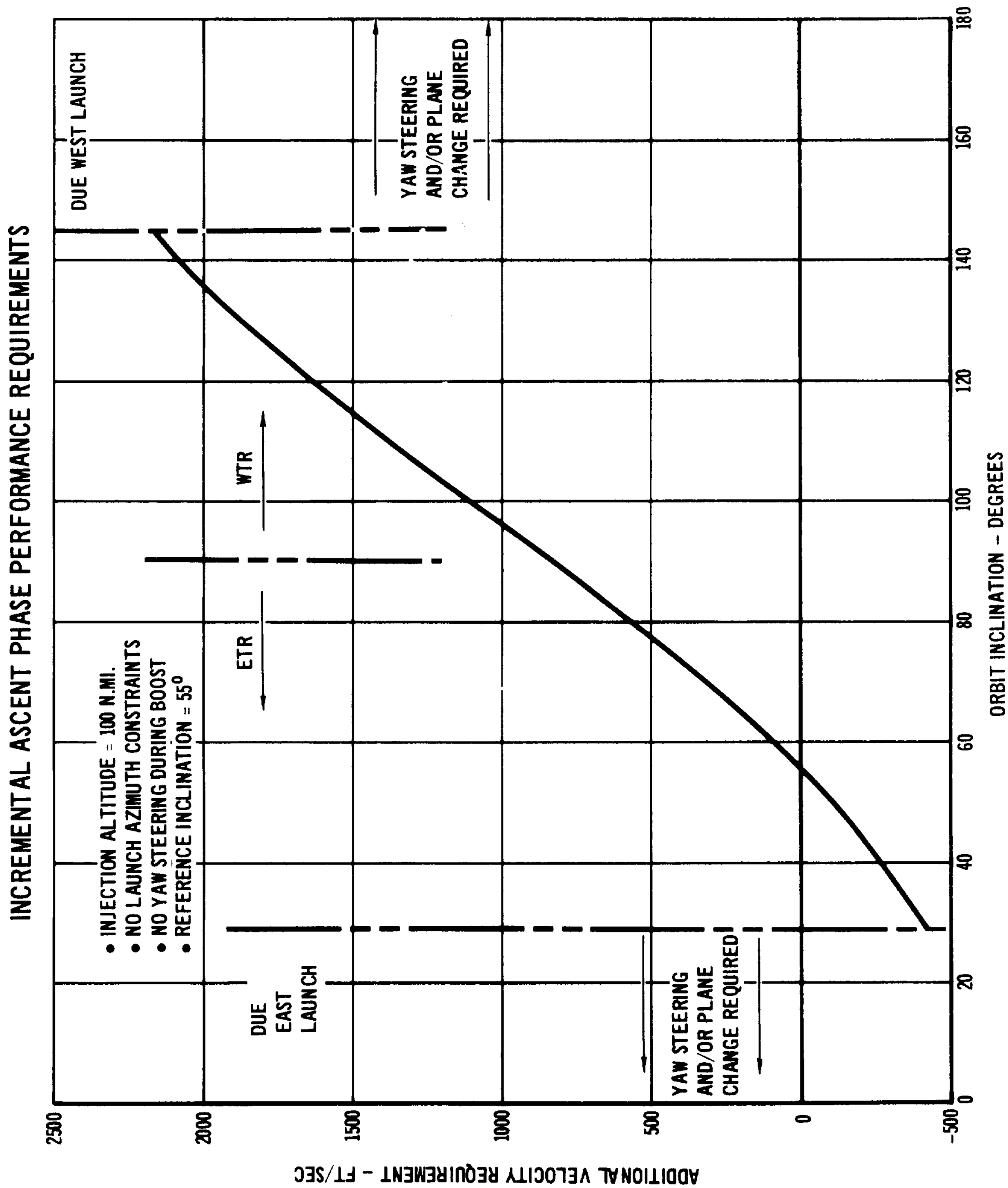

INCREMENTAL ASCENT PHASE PERFORMANCE REQUIREMENTS
• INJECTION ALTITUDE = 100 N.MI.
• NO LAUNCH AZIMUTH CONSTRAINTS
• NO YAW STEERING DURING BOOST
• REFERENCE INCLINATION = 55°
DUE WEST LAUNCH
YAW STEERING AND/OR PLANE CHANGE REQUIRED
WTR
ETR
DUE EAST LAUNCH
YAW STEERING AND/OR PLANE CHANGE REQUIRED
2500
2000
1500
1000
500
0
-500
ADDITIONAL VELOCITY REQUIREMENT - FT/SEC
0
20
40
60
80
100
120
140
160
180
ORBIT INCLINATION - DEGREES

Figure 2-3

HOHMANN TRANSFER ΔV REQUIREMENTS

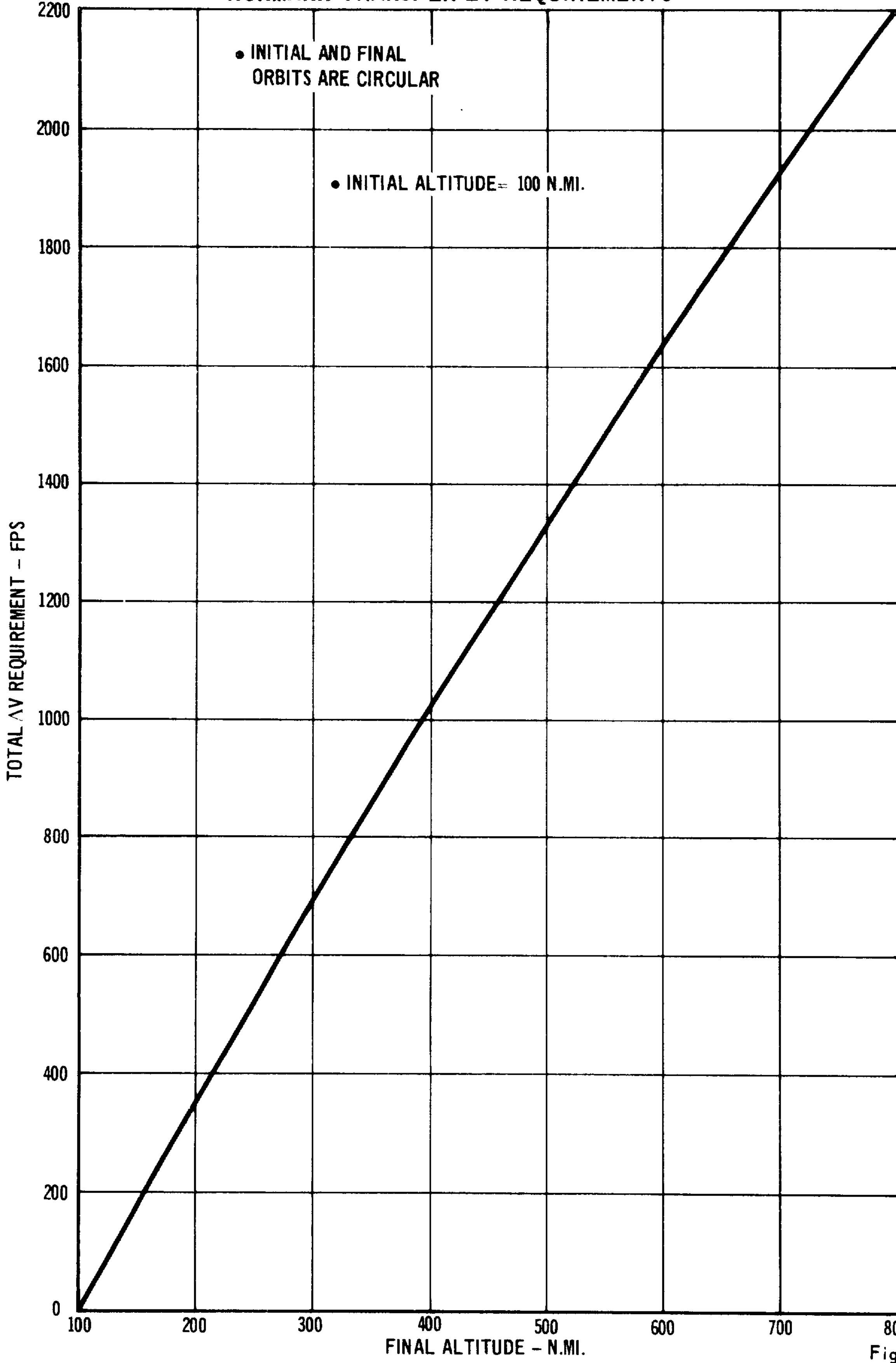


Figure 2–4

payload case has a working impulsive-velocity increment of 590 fps which corresponds to a 270 NM mission from Figure 2-4.

Next, additional values of impulsive velocity are determined from Figure 2-3 for various inclinations. These last values are subtracted from each of the constant payload values. Now, using Figure 2-4, the final altitudes can be determined with knowledge of the total impulsive-velocity requirements. Hence, plots of mission altitude versus mission inclination for lines of constant payloads can be drawn. This is illustrated in final form for the baseline vehicle in Figure 2-5.

Alternate-mission capability can be assessed if altitude and inclination ranges, together with minimum delivered payload, are specfied. Since mission altitude and inclination ranges combine to form rectangular plots on the chart of Figure 2-5, each of the prospective missions can be overlayed on the basic plots of constant-payload capability. Consequently, that area which falls below and to the left of the constant-payload line corresponding to the minimum mission payload, represents that portion of the overall mission which the baseline system can accomplish.

Typical mission requirements for a set of likely alternate missions are shown in Figure 2-5, where the circled numerals located in the rectangular areas correspond to the missions noted above by that numeral. An assessment as to the capability of the baseline system to accomplish these missions is given in Table 2-7 As indicated in the table, a range of 45.5 to 100 percent over all the missions can be attained by the baseline concept. For the propellant delivery missions, where 50,000 lbs of propellant is the payload requirement, it takes two trips by the baseline vehicle to accomplish 90 percent of that mission's inclination-altitude requirements.

There are some limitations to the above technique which are noted below:

- o No account is taken for fluctuations in deorbit velocity with varying mission altitude.
- o The percent-of-mission-covered quantities quoted assume an equal likelihood for all points within the altitude-inclination mission rectangle. This may not be the case. Certain altitude-inclination regimes may be more probable than others.

ALTERNATE MISSION CAPABILITY

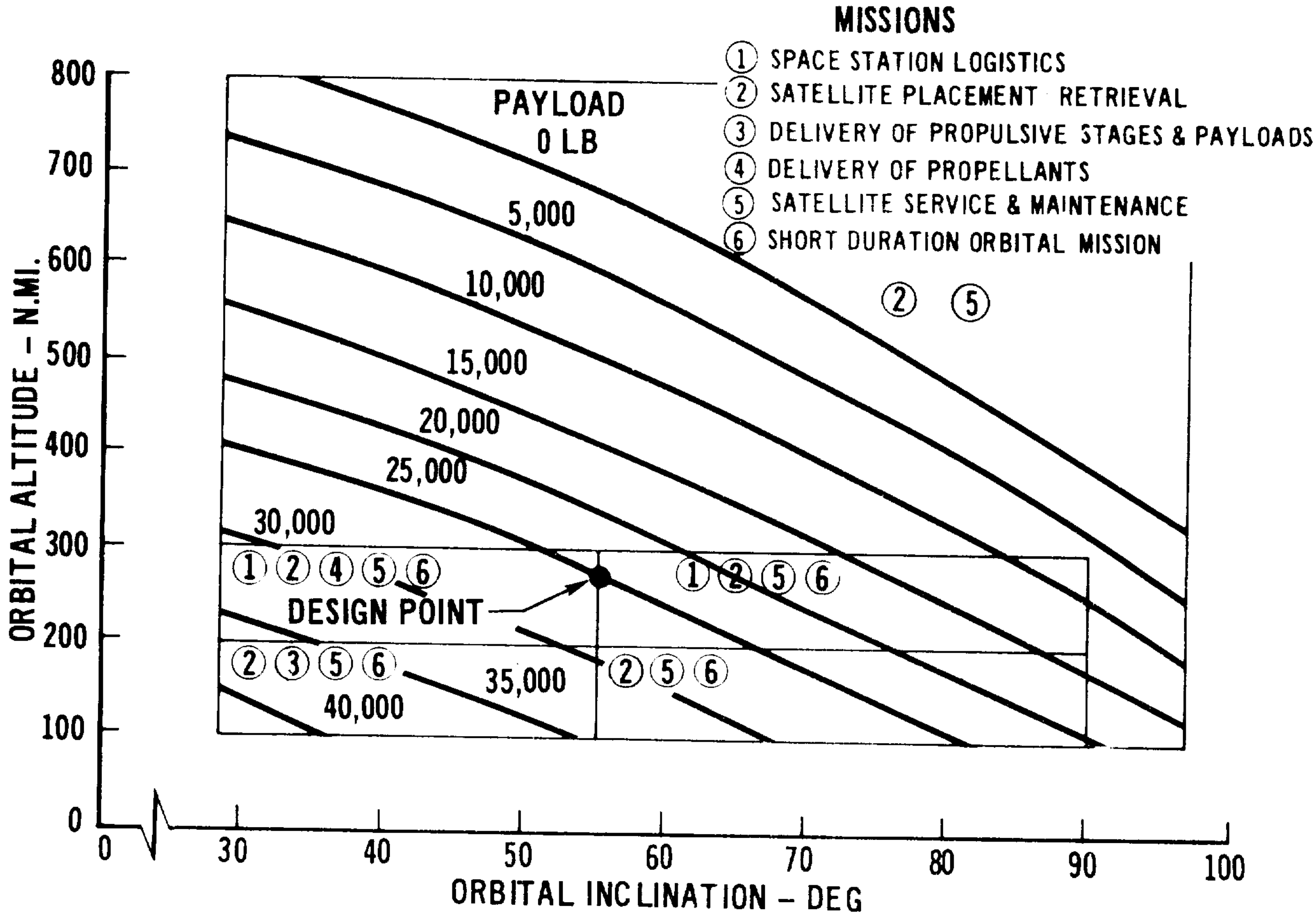


Figure 2-5

Table 2-7

MISSION PERFORMANCE
(Design Payload ≈ 25,000 Pounds)

	MINIMUM PAYLOAD/FLIGHT (LB)	PERCENT OF MISSION COVERED (%)
1. SPACE STATION LOGISTICS	25,000	49.2
2. SATELLITE PLACEMENT/RETRIEVAL	10,000	45.5
3. DELIVERY OF PROPULSIVE STAGES & PAYLOAD	25,000	100.0
4. DELIVERY OF PROPELLANTS	50,000	NOTE 1
5. SATELLITE SERVICE & MAINTENANCE	5,000	55.0
6. SHORT DURATION ORBITAL MISSION	25,000	61.2

1. IN ONE FLIGHT CONCEPT CAN SUPPLY 50% OF PAYLOAD TO 90% OF THE MISSIONS.

No attempt has been made to assess these methodology limitations but for assessments of mission capability, the technique described is both quick and reasonably accurate.

2.5.3 Alternate Mission Design Impacts - Since the payload has been defined as a cylindrical integral mission canister, the payload may be considered as a standard, self-contained module which, except for the mechanical connections (and, perhaps, some mission operations), operates independently from the Orbiter. Alternate missions, then, in this respect, would have very little impact on the design of the baseline vehicle.

From the data (percent-of-mission-covered) presented previously in Table 2-7, it is seen that the baseline vehicle can perform about half the logistics missions within the inclination and altitude ranges of 28-90 degrees and 200-300 NM, respectively. Further, the mission for the delivery of propulsive stages and payload can be completely accomplished with the presently defined vehicle. The propellant-delivery mission can be completely performed by resorting to multiple launches of the smaller payload. However, the percent-of-mission covered for the remaining alternate missions ranges from 45.5 to 61.2 percent.

To provide the baseline vehicle with a 100% alternate-mission capability, the vehicle would have to be initially designed to carry a much larger payload into its reference orbit. Then, payload capability could be traded for additional onboard propellant necessary for the spacecraft to accomplish the higher altitude and inclination missions. No assessment was made as to what value of deliverable payload would be required in order to completely perform all the alternate missions. Such a task would require a much better definition of the alternate missions under consideration and is beyond the scope of this study.

2.6 Inland Launch and Landing Sites - Launch and landing from inland bases can provide improved mission operational capability. An analysis of the spectrum of alternate missions indicates that polar launches (required in order to achieve total coverage of the earth's surface), such as those needed for the Earth Resources Satellites, would frequently be required to accomplish these missions. Polar launches are presently performed from the Western Test Range (WTR) by launching in a southerly direction. The WTR site is satisfactory for expendable launch vehicles, but, like at ETR, the use of recoverable boost stages poses serious operational problems for recovery and reuse, particularly because of the high probability of salt-water immersion in such cases. These problems and limitations can be overcome by use of continental inland bases for launch and recovery. Additional problems are associated with launching over populated areas, but these problems are not unknown, unexpected, or insolvable. Solutions to overland flights may be easier to find than means to effect a water recovery of the Carrier.

One of the principal factors affecting the choice of an inland launch site is the consideration of recovery operations. For example, a site in North Texas was initially considered as a candidate site, but proved suitable only for northerly launches, because launches to the south precluded any possibility of land recovery without seriously penalizing the Carrier.

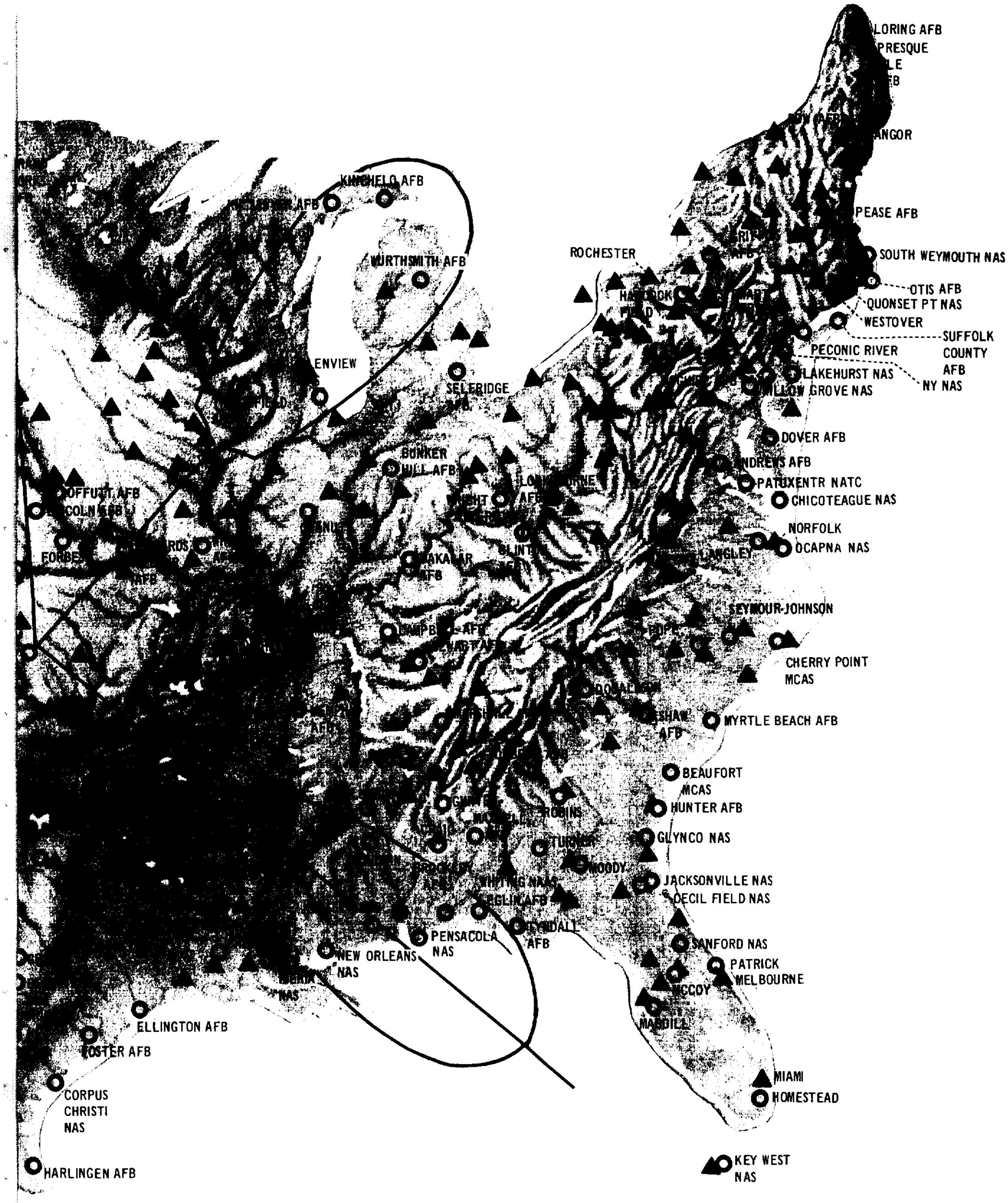
McConnell AFB, located just outside Wichita, Kansas, was selected as a most favorable inland launch-site candidate. Its location is such that no serious geographic constraints are encountered (mountains, lakes, deserts, etc.). Polar and other highly inclined launches are feasible from McConnell. In general, population densities are lower in the plain states than along either seaboard; no serious population shifts to the Midwest are anticipated. Also, there are a sufficient number of airports and USAF bases under the most-probable flight paths from McConnell to ensure adequate recovery capability under either a normal or abort operating mode.

A launch from McConnell AFB, Kansas, would occur along either a 47-degree or 133-degree azimuth to reach the baseline-mission inclination, 55 degrees.

The location of major civil and military airfields within the continential limits of the United States are shown in Figure 2-6. From the figure, it is seen that the topography of the western states is, in general, very rugged. Launches

INLAND LAUNCH AND LANDING SITES

FOLDOUT FRAME /



FOLDOUT FRAME 2

from California would be extremely constrained because of the mountain ranges extending for a considerable distance to the east and the Pacific Ocean bordering on the west. Easterly launches would very likely be required because of the increased payload capability of a posigrade launch in comparison to a retrograde (westerly) launch.

If launch azimuths were restricted to lie in the first quadrant (0° to 90°), then a land site in North-Central Texas, such as Sheppard AFB, would prove very attractive, in that weather extremes would be avoided, and heavily travelled air lanes would not pose a hazard. Also, with such azimuth constraints, abort-mode-recovery capability would be adequate from Sheppard AFB. However, the selection of such a southern location would require a larger "footprint" on the part of the Carrier for southerly launches, because it would be necessary to overfly non-US territories, or else to make dog-leg launches to avoid such overflights and still return to a friendly base.

Further studies, paricularly of climatic conditions, may indicate the advisability of an entirely new facility, optimally located for the ILRVS program.

3.0 AERO-THERMO PERFORMANCE ANALYSIS

This section considers the overall performance of the two stage system in each mission phase including launch, orbit, entry and landing. The basic groundrules underlying these performance analyses are summarized in Table 3-1. These groundrules were established through joint NASA/MDAC agreement. Most of the groundrules were NASA requirements while some were established on the basis of trade studies as indicated in Table 3-1. Sensitivites to these groundrules were assessed and the impact of variation in these parameters on vehicle size, weight and cost was determined. These sensitivities are presented throughout the report in the specific sections dealing with the parameter of interest e.g. weights, design and cost.

A summary of the primary performance characteristics of the two stage system based on the nominal groundrules defined in Table 3-1, is presented in Table 3-2. A brief discussion of these performance parameters is given in the following paragraphs.

Impulsive Velocity - The launch impulsive velocity split (ideal) between the first and second stage vehicles is determined primarily by the volume available in the 107 ft orbiter. The maximum velocity increment that can be incorporated in the orbiter is 16,777 ft/sec so the remainder (14,420 ft/sec) of the total launch velocity requirement effectively establishes the size of the carrier vehicle. An orbit velocity increment of 2000 ft/sec is provided in the orbiter for orbit maneuvering and attitude control.

Thrust-to-Weight - The lift off thrust-to-weight ratio of the Carrier is approximately 1.32 with all 10 engines functioning. With an engine out at lift-off this same thrust-to-weight can be maintained using engine overspeed. The initial thrust-to-weight ratio of the Orbiter at staging altitude is 1.42 with both engines functioning. With one engine out and with 25% engine overspeed, the initial orbiter thrust-to-weight is .89. An additional 590 fps of ΔV is required when operating in the engine out mode. However, the launch and an orbit ΔV contingencies will satisfy this requirement and enable the completion of the mission.

Maximum Acceleration - The launch trajectory for the two stage system as well as the entry trajectories for both stages were shaped so as not to exceed a 3g boundary. During launch the 3.g limit is maintained by a combination of throttling and shut-down of Carrier engines, and strict throttling of the Orbiter engines. During entry both vehicles approach but do not reach the 3.g limit.

Table 3-1

GROUNDRULE SUMMARY

- 25,000 lb. Cargo (Up and Down)
- 15' D x 30"L Cargo Container
- 10% Inert Weight Contingency (Both Stages)
- Series Burn
- Engine Out Capability (Both Stages, All Propulsion Systems)
- Flight Performance Reserve - 0.75% ΔV_{Boost}
- 2000 FPS On-Orbit ΔV
- ETR Launch
- 45 x 100 N Mi Injection Orbit at 55 Degree Inclination
- 3g's Maximum Acceleration (Eyeballs In & Down)
- * Carrier Cruise Back to Launch Site
- Orbiter Landing Go-Around Capability (10 Minutes Power)

* Selected From Trade Study Results

Table 3-2

VEHICLE PERFORMANCE SUMMARY

PARAMETER	CARRIER	ORBITER
Impulsive Velocity		
° Launch	14,420 fps (Ideal	16,777 fps (Ideal
° Orbital	-	2,000 fps
Initial T/W Boost	1.32	1.42 (Nominal
Maximum Acceleration	3 g	3 g
Propellant Fraction	.808 (Boost)	.684 (Boost)
Entry		
° Nominal Trajectory	Min. Down Range	Once/Day Return (390 NM CR)
° Wing Loading	38 psf	48 psf
° Max. Body Temp. (Aft of X/L = .125)	1100°F	2200°F
Cruise Range	357 n.m plus appr. and landing and 20% contingency	Appr. and Landing
Landing		
° Wing Loading (W/S)	33 psf	45 psf
° Touchdown Speed	133 Knots	164 Knots
° Touchdown	12°	23°

Propellant Fraction - Propellant fraction for the Carrier and Orbiter is defined to be the ratio of usable launch propellant to total stage weight at lift-off and staging respectively.

Entry Parameters - The nominal entry trajectory for the Carrier is characterized by a 180° inverted turn and maintaining a high angle of attack throughout most of the entry. This trajectory tends to minimize the Carrier down range and thus cruise requirements. The entry wing loading for the Carrier is 38 psf based on the total projected plan area (13,300 ft^2). The maximum body temperature that is experienced during entry (aft of the 12.5% body station) is 1100°F. The nose, fin leading edges, and flaps which comprise a small percentage of the total area realize somewhat higher temperatures.

The nominal entry trajectory for the Orbiter is characterized by a high angle-of-attack entry and approximately 390 NM cross range which provides once per day return capability at the 55° inclination reference orbit. The wing loading for the Orbiter at entry is 48 psf based on the total projected plan area (4,160 ft^2). The trajectory was shaped so as not to exceed a 2200°F heating boundary (aft of 12.5% body station).

Cruise Range - The Carrier has a cruise range of 357 NM plus approach and landing capabilities with an additional 20% contingency for head winds and hot day operation. This range is sufficient for the Carrier to return to the launch site. The Orbiter with once per day return capability has no cruise requirement but has propellant sufficient for go-around and a powered landing.

Landing Parameters - The wing loadings for the Carrier and Orbiter at landing are based on the total projected plan areas which are 13,000 ft^2 and 4,160 ft^2 respectively. The touchdown velocities are based on the touchdown angles defined in Table 3-2. The values of these touchdown angles are limited by the landing gear design and vehicle geometry.

The following sections present the detailed performance analyses by mission phase and include two special emphases areas, Approach and Landing, and Abort.

3.1 Ascent Performance - The primary ascent performance study effort was addressed to trajectory shaping, stage separation dynamics, launch configuration aerodynamic characteristics, and heating. In the area of trajectory shaping, trade studies were conducted to determine velocity loss sensitivities to key system variables. The separation analysis resulted in the definition of a viable separation sequence. Aerodynamic characteristics were derived from exploratory wind tunnel tests of the launch configuration conducted at the Langly Research Center test facilities. The heating analysis provided a comparison of launch and reentry temperatures for the Carrier and Orbiter.

3.1.1 Ascent Phase Aerodynamic Characteristics - The subsonic lift, drag, and moment characteristics for the Ascent Phase configuration are presented in Figures 3-1 through 3-4. The information shown here was obtained from the results of an exploratory wind tunned test conducted at the Langley Research Center Low Turbulence Pressure Tunnel. The force coefficients were normalized with a reference area which corresponded to the theroretical wing area of the carrier vehicle, while the moments were normalized with the same reference area and the corresponding mean aerodynamic chord. The moment reference point was positioned at the 45% station on the carrier, which is representative of the center of gravity location of the combined masses of the loaded carrier and orbiter.

The lift data show that the configuration has a zero angle-of-attack lift coefficient of 0.21 and a lift curve slope of 0.042 per degree for the angle-of-attack range from zero to eight degrees. Comparison of these ascent configuration data with the carrier test data alone (presented in succeeding paragraphs) shows that the carrier lift characteristics are almost the same, indicating that the carrier lifting forces predominate, at least over the positive angle-of-attack range.

Comparison of the drag data shows that the zero angle-of-attack drag coefficient of 0.073 is twice the value for the carrier alone, which indicates that there is a sizeable interference factor present. The subsonic estimates for zero angle-of-attack drag did not include an interference factor, and therefore underpredicted the drag by a sizeable margin. However, it must be pointed out that the test data did not include base pressure corrections due to engine thrust effects and may still not be a true indicator of the actual boost phase drag. Also shown on the zero angle-of-attack drag figure is the Mach number range where the majority of the drag losses are accumulated during a nominal ascent trajectory. The transonic drag region is delineated as the area which must be well defined for accurate drag loss predictions.

The subsonic moment curve indicates that the ascent configuration is very stable with respect to the chosen moment reference point. A combined center-of-gravity location as far aft as the 59.2% station on the carrier could be tolerated before a neutral stability condition would exist. This large static margin is due mainly to the large wing of the carrier, but the drag moment of the orbiter introduces a large nose-down moment which is not indicated by

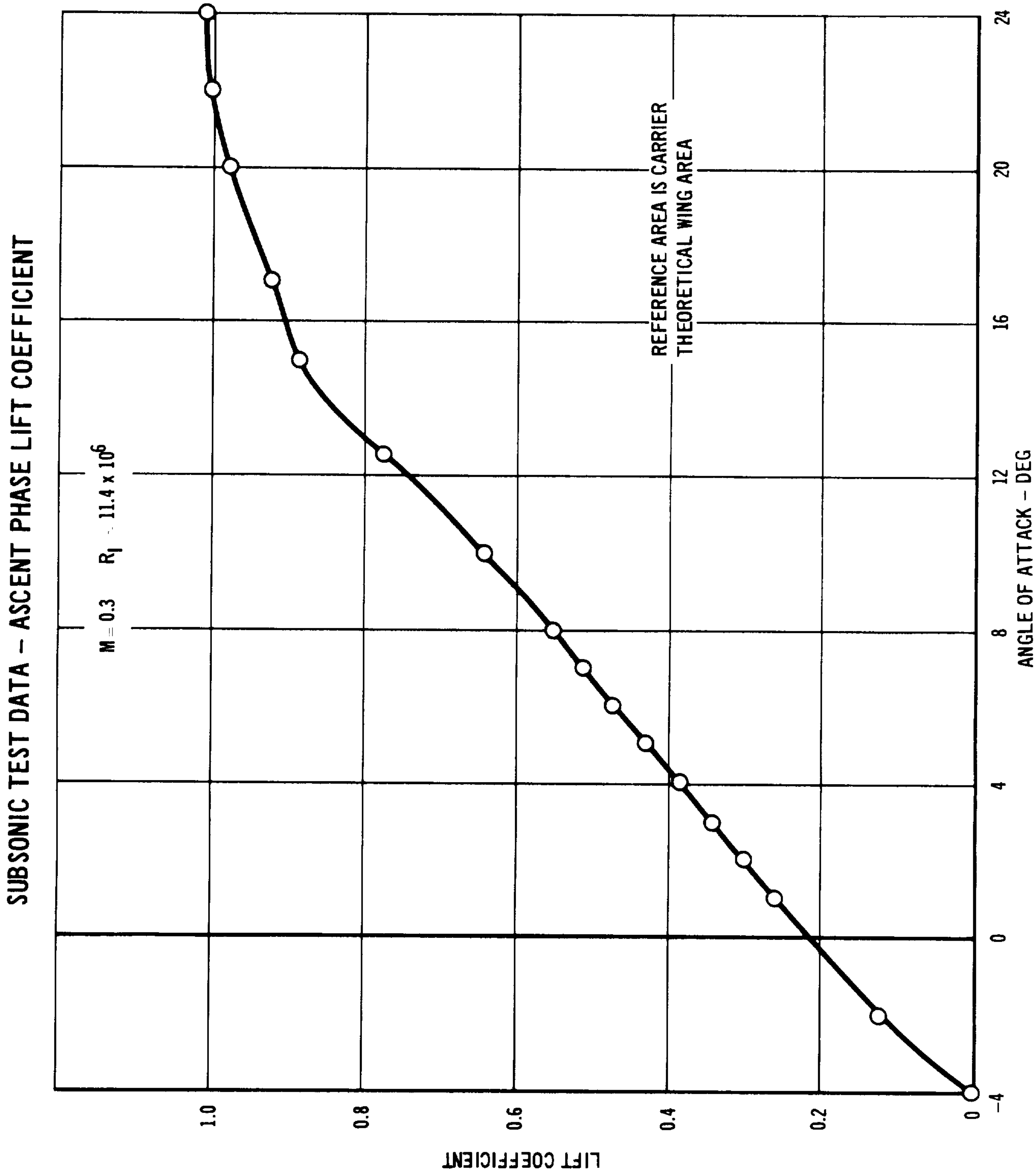

SUBSONIC TEST DATA – ASCENT PHASE LIFT COEFFICIENT
M = 0.3   R_l = 11.4 x 10^6
REFERENCE AREA IS CARRIER THEORETICAL WING AREA
LIFT COEFFICIENT
0
0.2
0.4
0.6
0.8
1.0
ANGLE OF ATTACK – DEG
-4
0
4
8
12
16
20
24

Figure 3-1

SUBSONIC TEST DATA – ASCENT PHASE DRAG COEFFICIENT

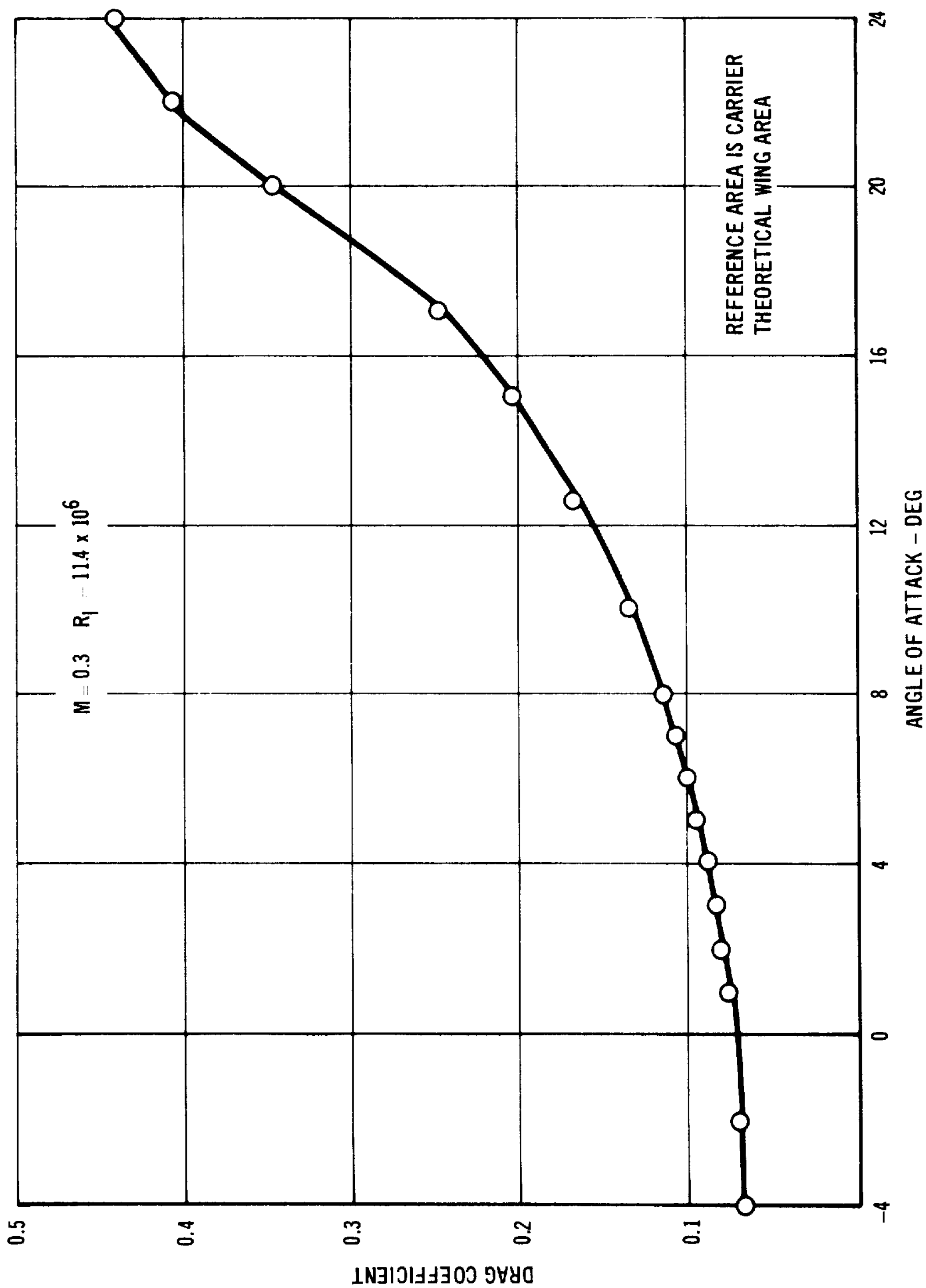


Figure 3–2

SUBSONIC TEST DATA

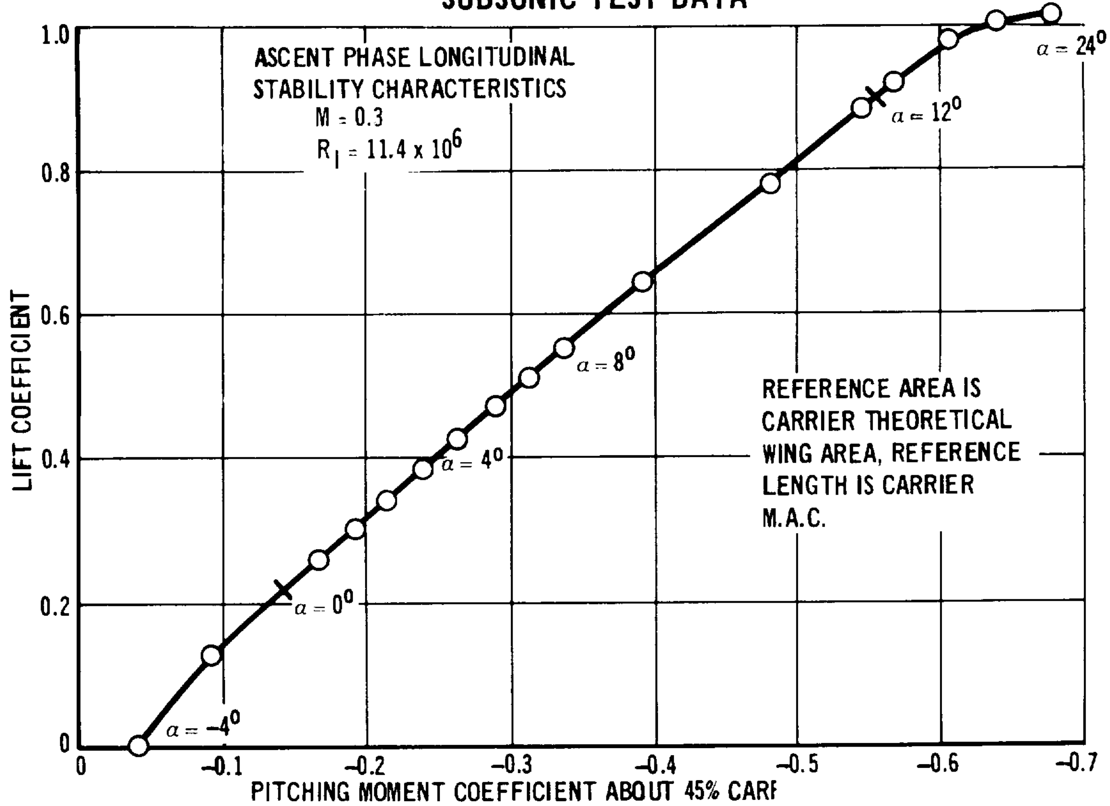


Figure 3–3

ASCENT CONFIGURATION DRAG COEFFICIENT

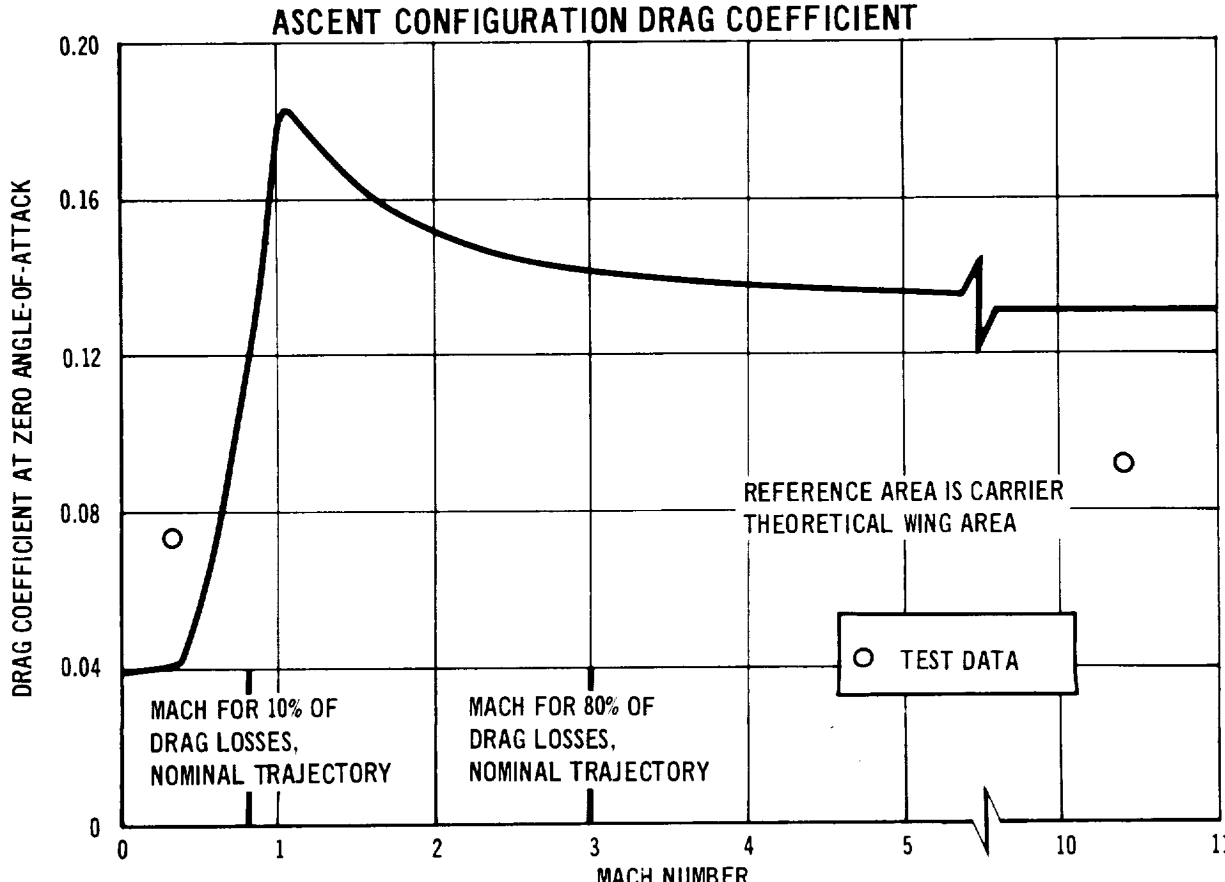


Figure 3–4

ILRVS–468 F

carrier-alone data. Additional elevon deflection data, if available, would show that it was possible to trim the configuration at low angle-of-attack. However, this would be true for the power-off condition only (abort mode), since the power-on trim conditions are drastically affected by thrust moments of the rocket engines.

Figures 3-5 through 3-7 present the hypersonic lift, drag, and moment characteristics of the ascent phase configuration. This information was obtained from the results of a wind tunnel test conducted at the Langley Research Center Continuous Flow Hypersonic Tunnel. The reference area and length is the same as that used for the subsonic data. The moment reference point was positioned at the 74% station on the carrier, which is a point approximately 10% aft of the center of gravity location of the combined masses at staging.

The lift figure shows that the configuration has a zero angle-of-attack lift coefficient of 0.025 and a lift curve slope of 0.012 per degree. Comparison with the carrier hypersonic data indicates that the zero angle value is different in sign, and the lift curve slope of the ascent phase configuration is about 20% higher than that from the carrier alone. The increase in lift effectiveness is believed to be due to the increased local pressure on the bottom of the carrier wing caused by the intersection of the shock wave from the orbiter vehicle.

Comparison of the drag data shows that the zero angle-of-attack drag coefficient of 0.091 for the ascent phase configuration is about 35% greater than the carrier drag alone. The numerical sum of the carrier drag (.066) and the orbiter drag (.034, based on carrier wing area) is greater than the drag of the ascent phase configuration, which implies that favorable interaction of the shock systems from the two vehicles is causing local pressure regions to reduce the component drags. No base pressure corrections were included to account for rocket thrust effects, but the power-on condition could change the drag appreciably.

The moment curve indicates that the ascent configuration is unstable with respect to the chosen moment reference point. However, the figure shows the addition of center-of-gravity reference lines and indicates that the vehicle will be at least neutrally stable with a nominal staging center-of-gravity; i.e. 65 to 66 percent of body length.

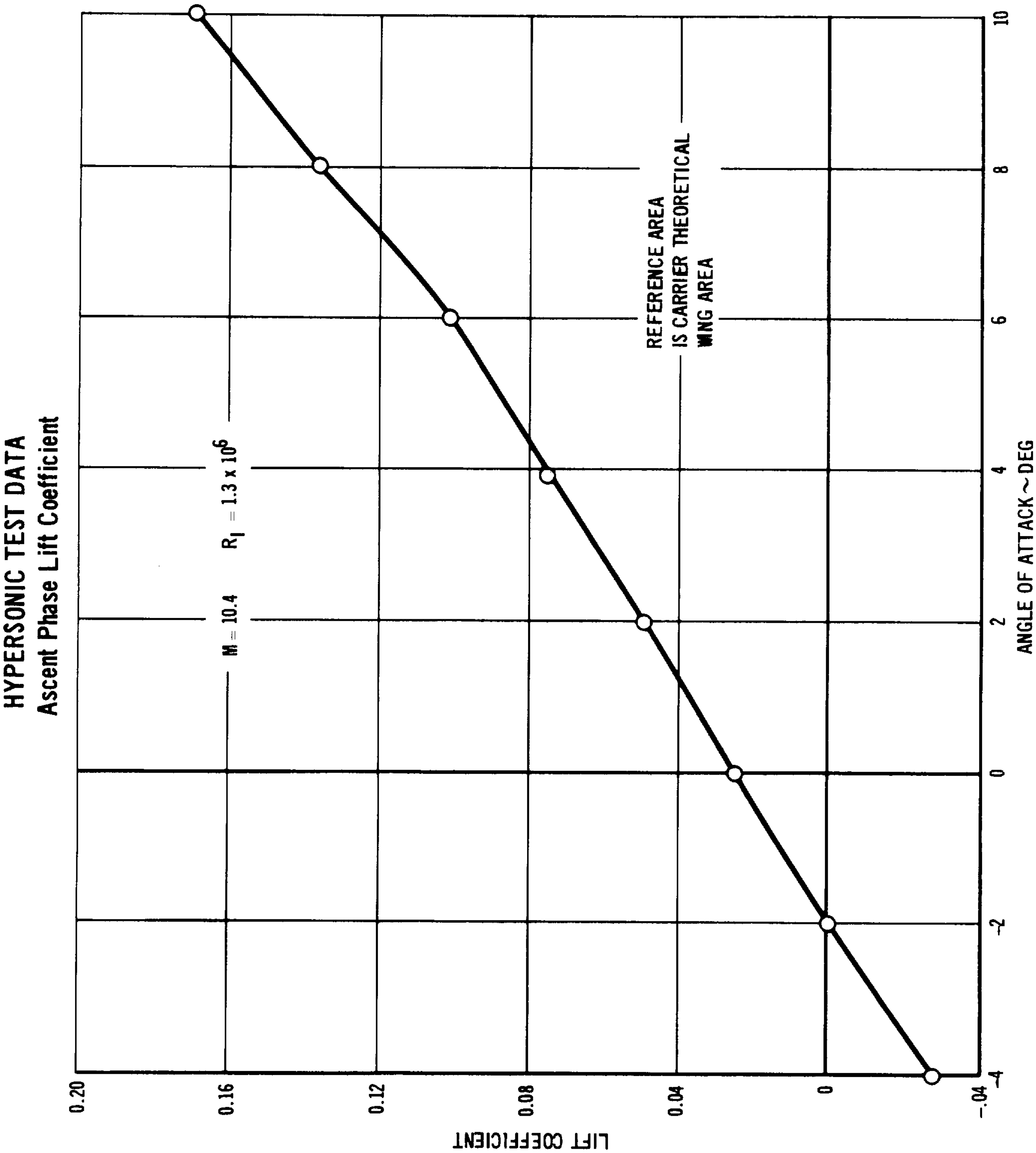

HYPERSONIC TEST DATA
Ascent Phase Lift Coefficient
M = 10.4
R_l = 1.3 x 10^6
REFERENCE AREA
IS CARRIER THEORETICAL
WING AREA
LIFT COEFFICIENT
0.20
0.16
0.12
0.08
0.04
0
-.04
ANGLE OF ATTACK ~ DEG
-4
-2
0
2
4
6
8
10

Figure 3-5

HYPERSONIC TEST DATA
Ascent Phase Drag Coefficient

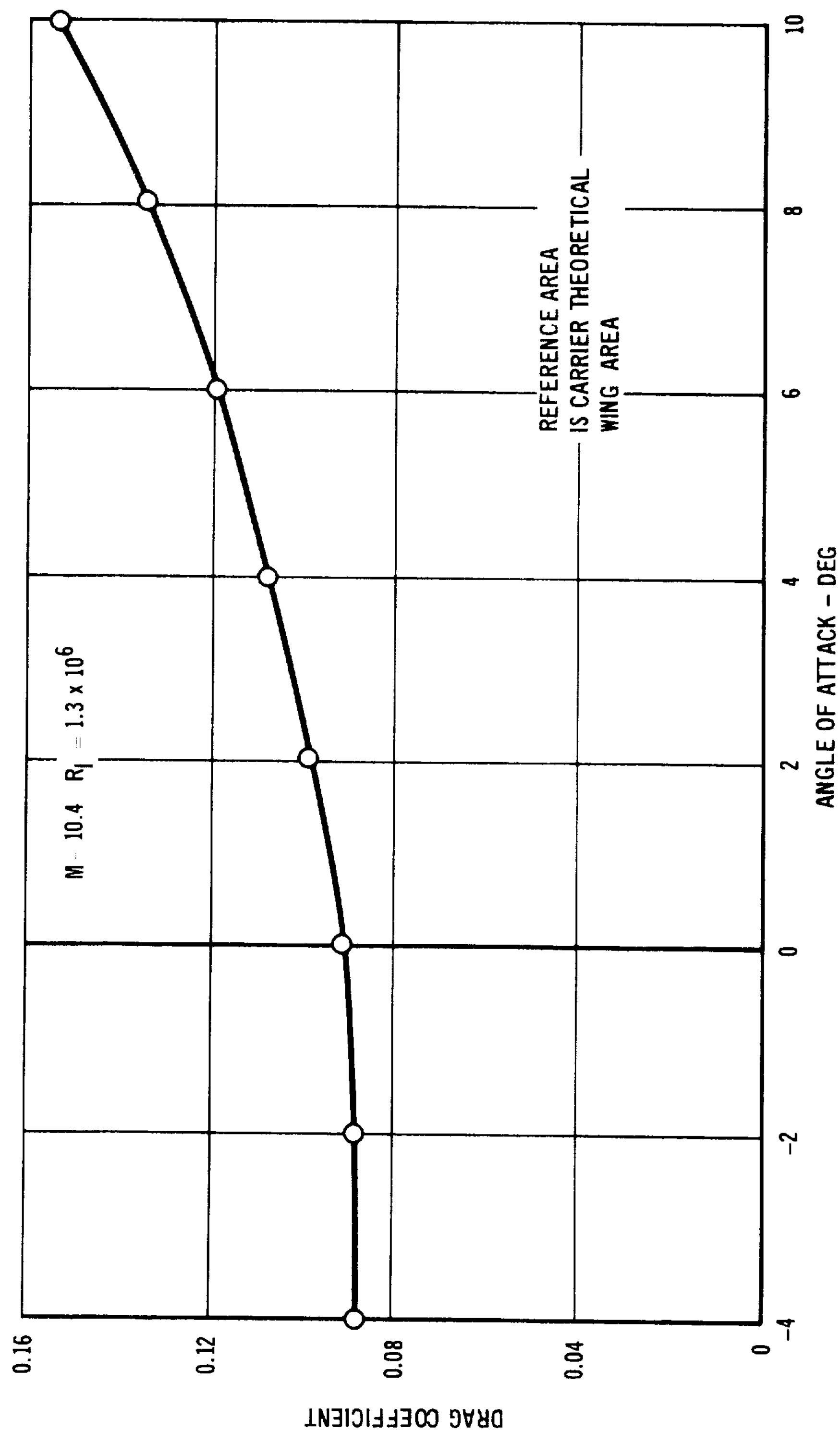


Figure 3-6

HYPERSONIC TEST DATA
Ascent Phase Longitudinal Stability Characteristics

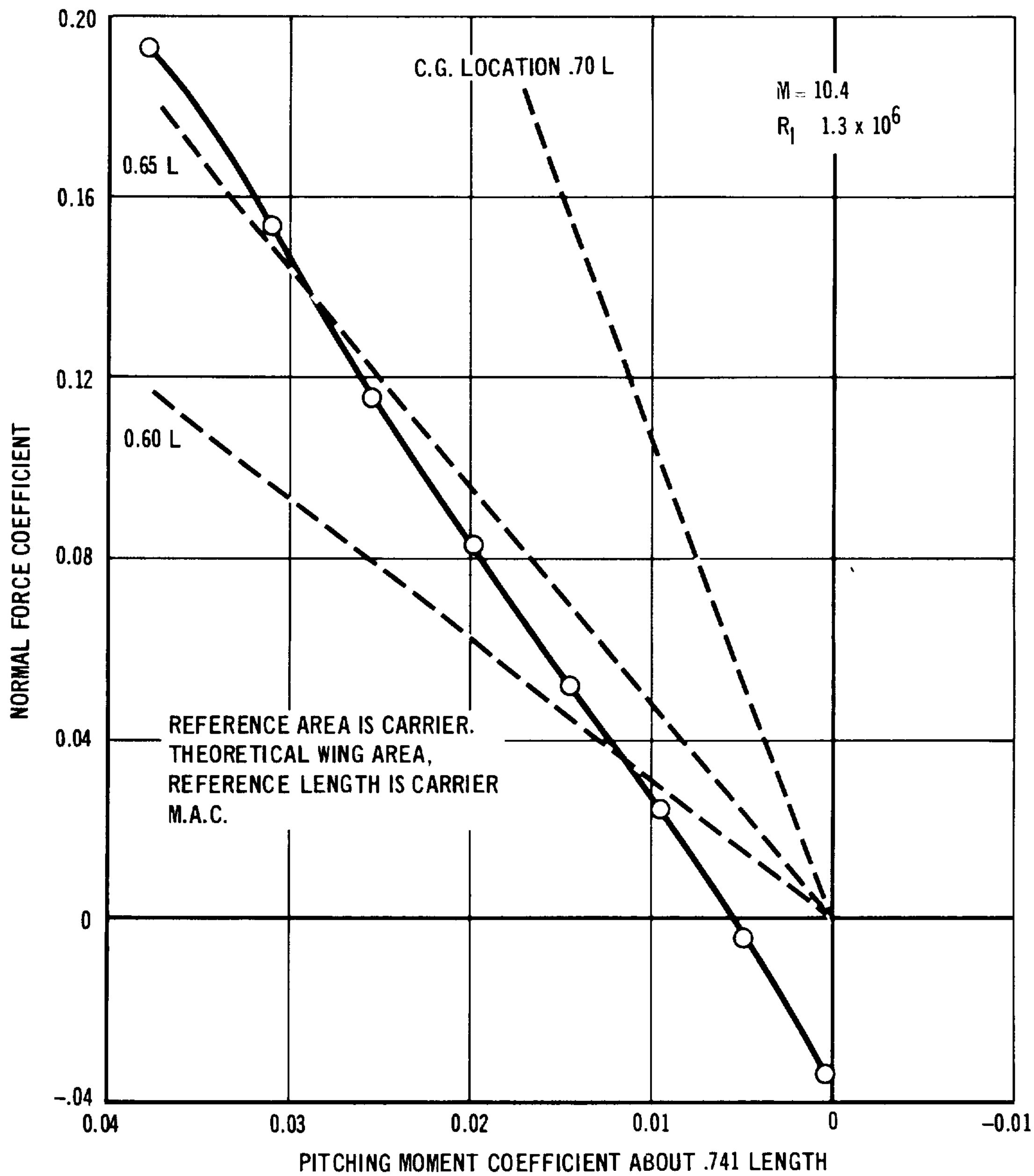


Figure 3-7

3.1.2 Ascent Trajectory Analysis - Ascent trajectory design is subject to a number of trade-offs. The final design is the result of a process which begins by identifying and assigning weighting values to the significant dependent variables and ends when the design is found which maximizes the payoff function(s) within prespecified groundrules. The objective of this section is to identify some of the key tradeoff values and assess their interaction on vehicle and trajectory design. Finally, key ascent trajectory environmental variables for the selected configuration baseline design are presented.

The total mission velocity budget that must be built into the launch configuration is the sum of (a) characteristic ideal mission velocity (including the earth rotational component), (b) nominal ascent phase losses, and (c) a contingency termed flight performance reserve. Since (a) and (c) have been specified, the primary objective of ascent trajectory shaping can be stated as minimizing velocity losses.

Velocity Loss Trades - Ascent phase losses comprise 15 to 20 percent of the mission design velocity budget. Gross launch weight sensitivity to velocity budget is high ($\approx$ 530 Lbs/Per Ft/Sec of Orbiter ΔV). It was therefore imperative to identify and assess the sensitivity of those performance parameters characterized by high velocity loss trades. Trade studies were performed to evaluate the effect on velocity losses of:

- Lift-off thrust-to-weight ratio
- Orbiter initial thrust-to-weight ratio
- Staging coast time
- Maximum axial load factor of 3G's and 4G's

Analytic calculations in most instances require assumptions and simplifications which would at best yield only order-of-magnitude answers. Consequently, numerical analysis techniques were incorporated wherein integrated ascent trajectories were run for several discrete values of the parameters. The trajectory program utilizes a rotating spherical earth model and the 1962 U.S. standard atmosphere. A gravity turn was simulated for the first stage and a thrust vectoring program derived by a calculus-of-variations scheme was used in the second stage to achieve the desired insertion conditions. Drag forces were simulated during first stage operation using the combined vehicle drag curve in Figure 3-4.

Lift-Off Thrust/Weight - Losses reduced monotonically with increasing F/W as shown in Figure 3-8. The exchange rate is approximately 330 ft/sec per 0.1 g at F/W = 1.317 g's. This figure does not reflect the compensating effect of increasing system weight with increasing engine size and/or number. Other considerations which restrict high F/W design selection include base area limitations and a requirement for the same engine in both stages.

Orbiter Thrust/Weight - The study requirement for engine out capability without mission compromise was a key factor in boost engine selection. Loss of one first stage engine can be compensated for with moderate over-speed of the remaining engines with no ΔV penalty. However, a second stage engine failure (for a two or three engine configuration) results in a severe ΔV penalty. The velocity loss curve in Figure 3-9 is characterized by a sharp increase in gravity and maneuver losses for F/W below about 1.2 g's. Single engine operation is in the region well beyond the "knee" of the curve, hence high losses are incurred. Without overspeed, losses are 1300 ft/sec greater than nominal two-engine operation. The loss increment is 820 ft/sec with 15% overspeed and 590 ft/sec with 25% overspeed. The velocity budget has been designed to accommodate a second stage engine out condition with the remaining operative engine running at 25% overspeed.

Staging Coast Time - A stage separation sequence is defined in part 3.1.3 of this section and is used here to identify and assess gross separation dynamics. Carrier thrust tail-off, orbiter pre-start chill-down requirements, thrust build-up history, etc. had not been defined in enough detail to warrant inclusion in the trajectory simulation. A trade study was performed, however, to determine losses as a function of coast time between stages. The assumptions were:

- Instantaneous thrust termination at staging.
- A free-fall unpowered coast interval, and
- Instantaneous thrust build-up at second stage engine ignition.

The exchange rate is 6.5 ft/sec loss per second of coast time across the 20 second coast interval examined.

EFFECT OF LIFT-OFF F/W ON VELOCITY LOSSES

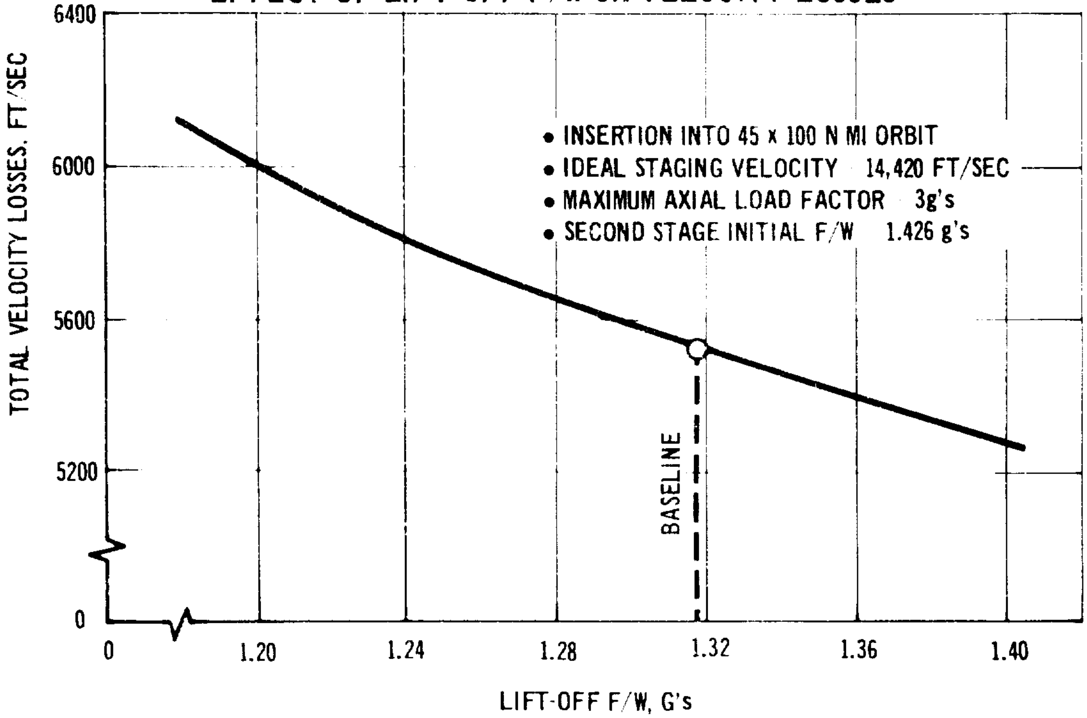


Figure 3-8

EFFECT OF SECOND STAGE INITIAL F/W ON VELOCITY LOSSES

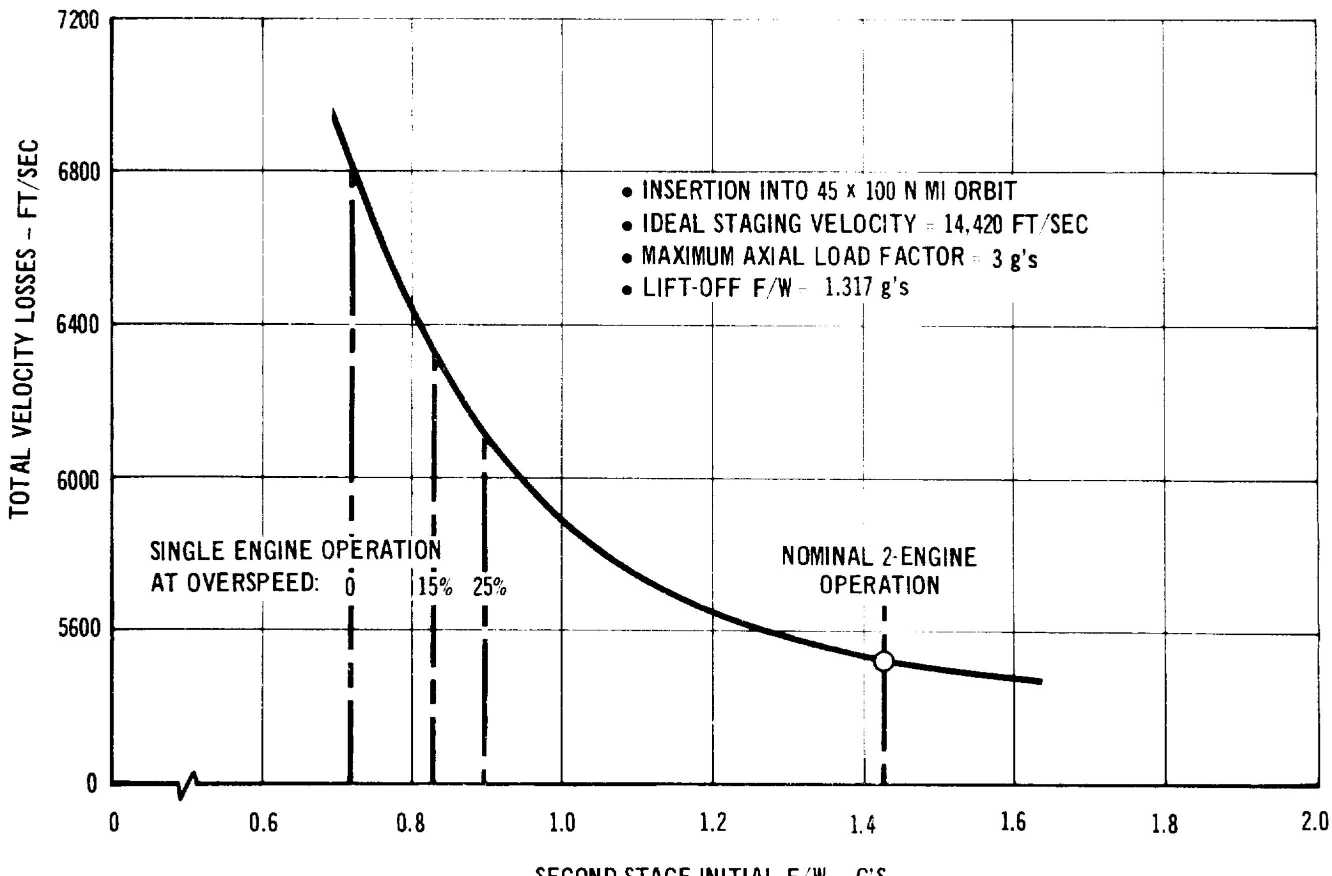


Figure 3-9

Maximum Axial Load Factor - The effect of increasing the permissible maximum axial load factor from 3 g's to 4 g's was to decrease losses and hence, the velocity budget by 40 ft/sec. Table 3-3 provides a breakdown of the total design velocity budget. Note that the gravity loss term constitutes approximately 77% of the total losses. The next largest contributor is drag at 14%.

Baseline Ascent Trajectory - From the performance trades outlined in this section and similar systems/performance trades described elsewhere, a baseline configuration was defined which satisfies the required mission and performance criteria. The key configuration and system performance characteristics required for ascent trajectory shaping are:

- Boost propulsion system (thrust and I_{SP})
- Weights (propellant, structure, and payload)
- Aerodynamic drag

The boost propulsion system is comprised of ten 448,000 lb sea level thrust engines in the first stage and two in the second. Propellant is loaded to provide 14,420 ft/sec ideal velocity in the first stage and the design velocity budget balance in the second stage. Stage operation is series burn, i.e. second stage burn is initiated following first stage shut-down and separation. The resultant ascent phase altitude thrust and propellant flow rate time histories are shown in Figure 3-10. At 48 seconds the first stage two position nozzles are extended to secure the higher specific impulse.

From a performance standpoint, the predominant aerodynamic force during ascent is drag. The drag curve (C_D vs Mach number) used is presented in Figure 3-4. Drag losses were determined to trade-off at approximately 4 ft/sec per percent change in drag coefficient over the transonic Mach range.

Mission Profile and Flight Sequencing - A typical in-plane mission flight profile is illustrated in Figure 3-11. The view is from a south westerly direction, normal to the polar orbital plane (55° inclination). Transfer from the 100 nm parking orbit to the 270 nm station altitude orbit is accomplished using the modified limited rev technique defined and discussed in Section 3.2 of this volume. Figure 3-12 shows the ascent phase and first stage recovery altitude/range profiles. Detailed second stage reentry and first stage recovery trajectory shaping is discussed in Section 3.3 of this volume.

Table 3-3

VELOCITY BUDGET SENSITIVITY
Nominal Performance

	MAXIMUM AXIAL LOAD FACTOR	
	3G'S	4G'S
• INSERTION VELOCITY (45 x 100 N MI, i = 55°)	25,000	25,000
GRAVITY LOSSES	4,283	4,263
MANEUVER LOSSES	145	129
ALTITUDE THRUST LOSSES	340	340
DRAG LOSSES	764	760
• NOMINAL ASCENT PHASE BUDGET, ΔV_N	30,532	30,492
FLIGHT PERFORMANCE RESERVE (0.75% ΔV_N)	229	229
• TOTAL ASCENT PHASE BUDGET	30,761	30,721
ON-ORBIT BUDGET	2,000	2,000
• TOTAL ΔV BUDGET	32,761	32,721

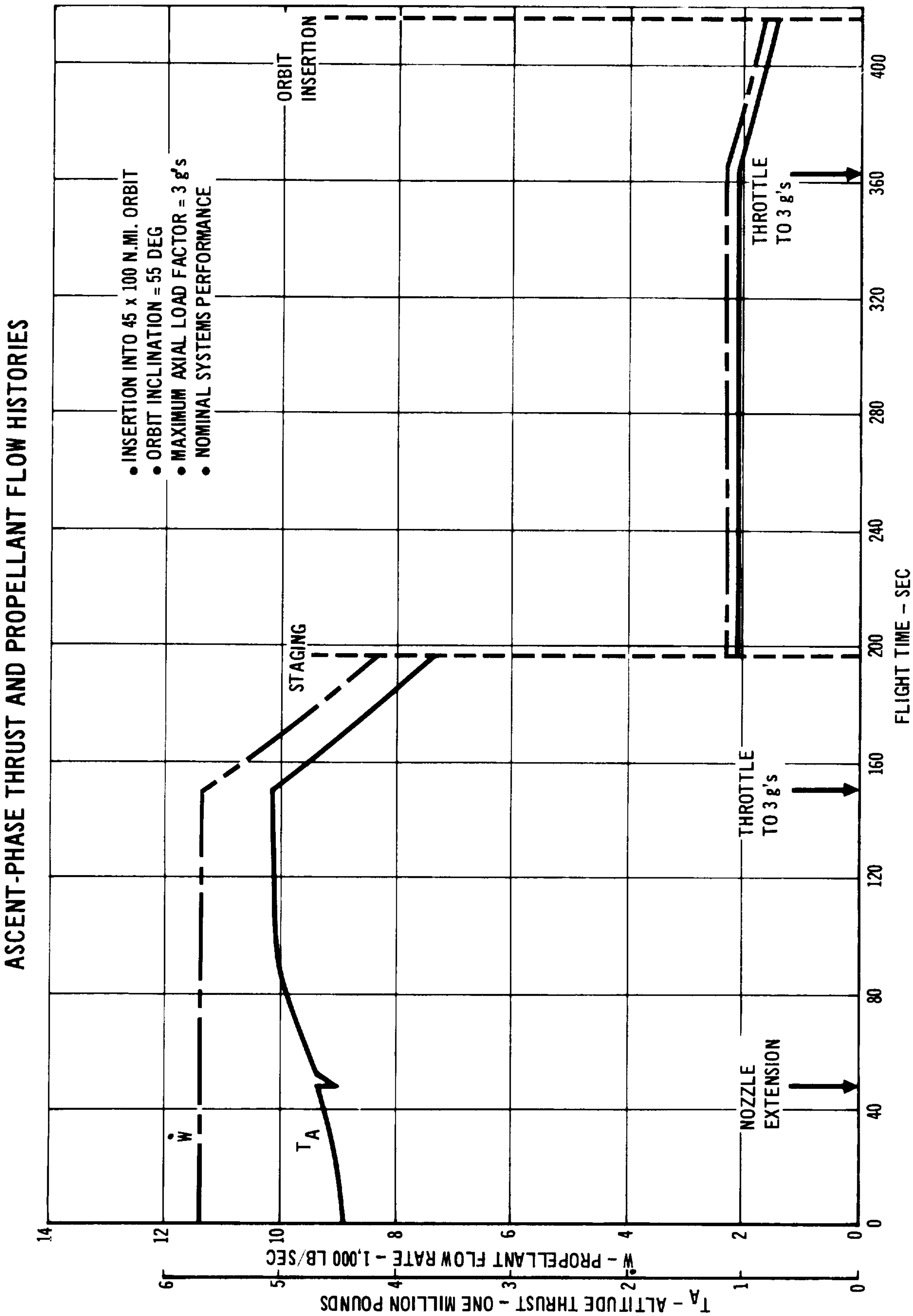
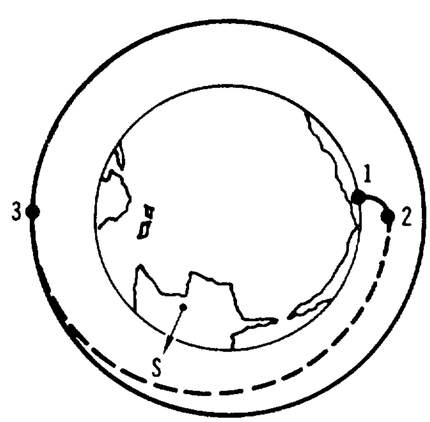

ASCENT-PHASE THRUST AND PROPELLANT FLOW HISTORIES
• INSERTION INTO 45 x 100 N.MI. ORBIT
• ORBIT INCLINATION = 55 DEG
• MAXIMUM AXIAL LOAD FACTOR = 3 g's
• NOMINAL SYSTEMS PERFORMANCE
ORBIT INSERTION
STAGING
THROTTLE TO 3 g's
THROTTLE TO 3 g's
NOZZLE EXTENSION
Ẇ
T_A
FLIGHT TIME – SEC
0
40
80
120
160
200
240
280
320
360
400
Ẇ – PROPELLANT FLOW RATE – 1,000 LB/SEC
0
2
4
6
8
10
12
14
T_A – ALTITUDE THRUST – ONE MILLION POUNDS
0
1
2
3
4
5
6

Figure 3-10

INPLANE MISSION PROFILE

ASCENT	t (MIN)	V (FPS)	h (NM)	R (NM)
1 LIFT-OFF	T_0	0	0	0
2 INSERTION	$T_0 + 7.0$	25,885	45	653
3 CIRCULARIZATION	$T_0 + 50.5$	25,583	100	11,463

REENTRY	t (MIN)	V (FPS)	h (NM)	R (NM)	τ (DEG)
1 RETROGRADE	$T_E - 31.5$	24,510	270	-7800	0
2 RE-ENTRY	T_E	25,990	65.8	0	-1.5
3 CRITICAL TEMP	$T_E + 6.3$	24,029	37.7	1570	≈0
4 LANDING	$T_E + 16.5$	284	0	2776	0



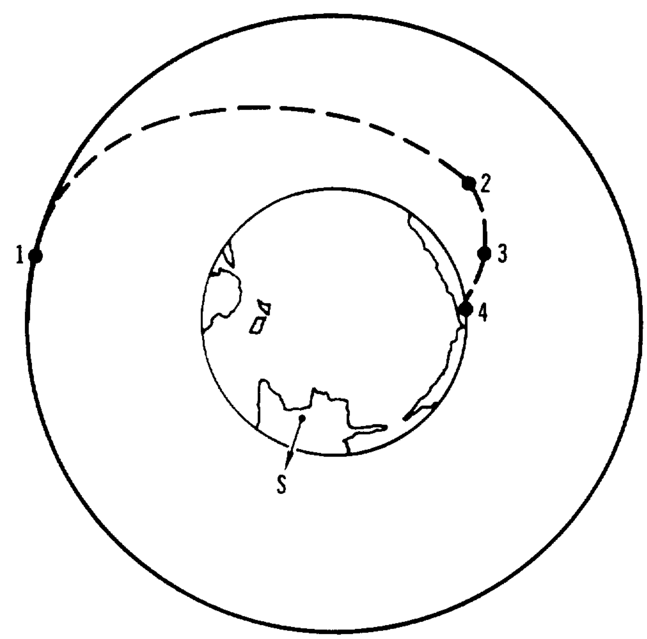
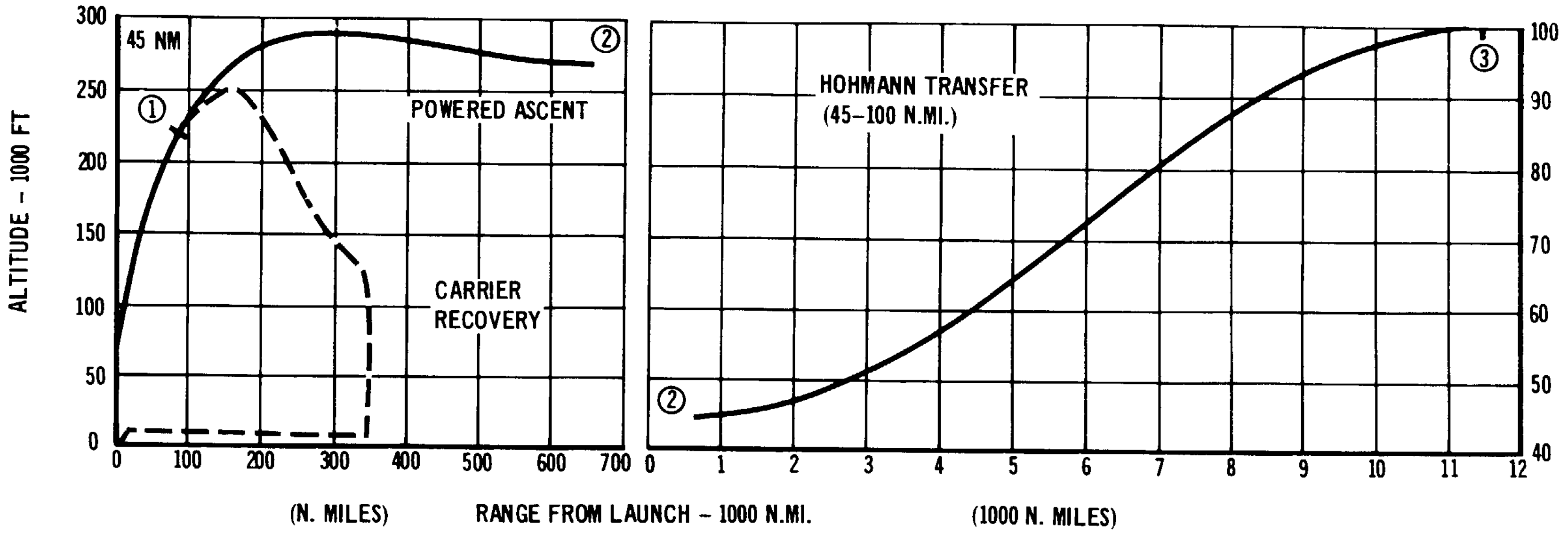


Figure 3-11

ASCENT-PHASE PROFILE



	t (MIN)	V (FT/SEC)	h (NM)	RANGE (NM)	α (DEG)
(1) STAGING	3.3	* 9240	36	83	*8.71
(2) INSERTION	7.0	25885	45	653	0
(3) CIRCULARIZATION	50.5	25583	100	11463	0

* RELATIVE

Figure 3-12

A nominal sequence of events and associated times are tabulated below. Beginning at lift-off, a programmed 20 second vertical rise and roll maneuver is executed. The roll program serves to rotate the launch configuration from the launch aligned azimuth to the desired flight azimuth.

EVENT	NOMINAL TIME (SEC)
o Lift-off	T
o Terminate vertical rise; initiate gravity turn	T + 20
o Extend nozzles from stowed position	T + 48
o Throttle to 3g axial load factor	T + 150.1
o Staging; initiate second stage thrust vectoring	T + 196.2
o Throttle to 3g axial load factor	T + 362.5
o Orbit insertion (45 x 100 NM); begin Hohman Transfer	T + 417.3
o Circularize at 100 NM	*T + 50.5 Min

Trajectory Parameters - Nominal ascent trajectory parameter time histories are shown in Figure 3-13 through 3-15. These data are intended to reflect typical trajectory characteristics. Maximum dynamic pressure was approximately 465 lbs/ft^2 and occurred 70 seconds following lift-off. An idealized gravity turn maneuver was simulated throughout the high q region to minimize aerodynamic loads. At staging, q had dropped to 10 lbs/ft^2. Note that the maximum load factor was maintained for about 45 seconds during first stage operation and for approximately 55 seconds during the second stage.

ASCENT-PHASE TRAJECTORY PARAMETERS (PART I)

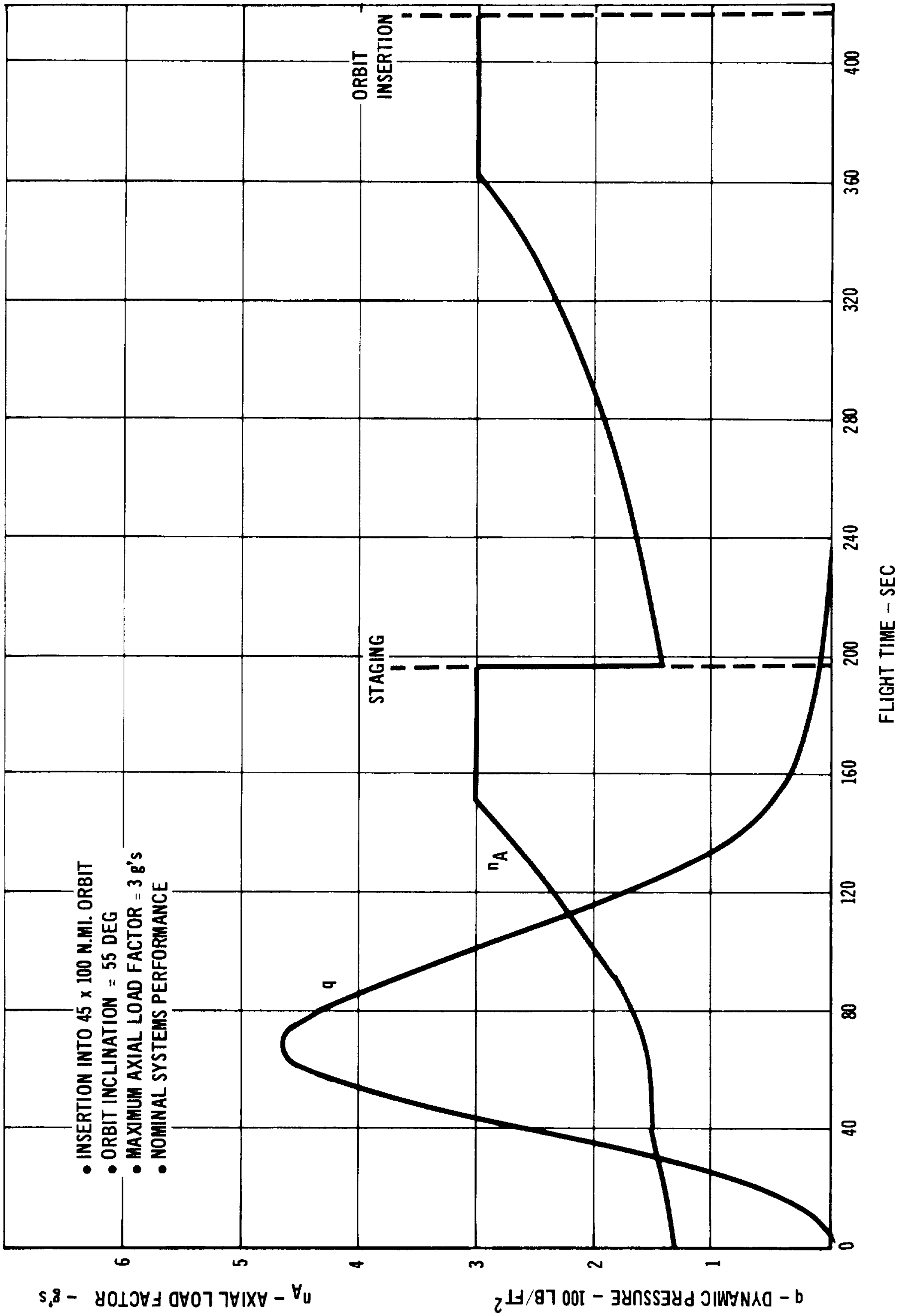


Figure 3–13

ASCENT-PHASE TRAJECTORY PARAMETERS (PART II)

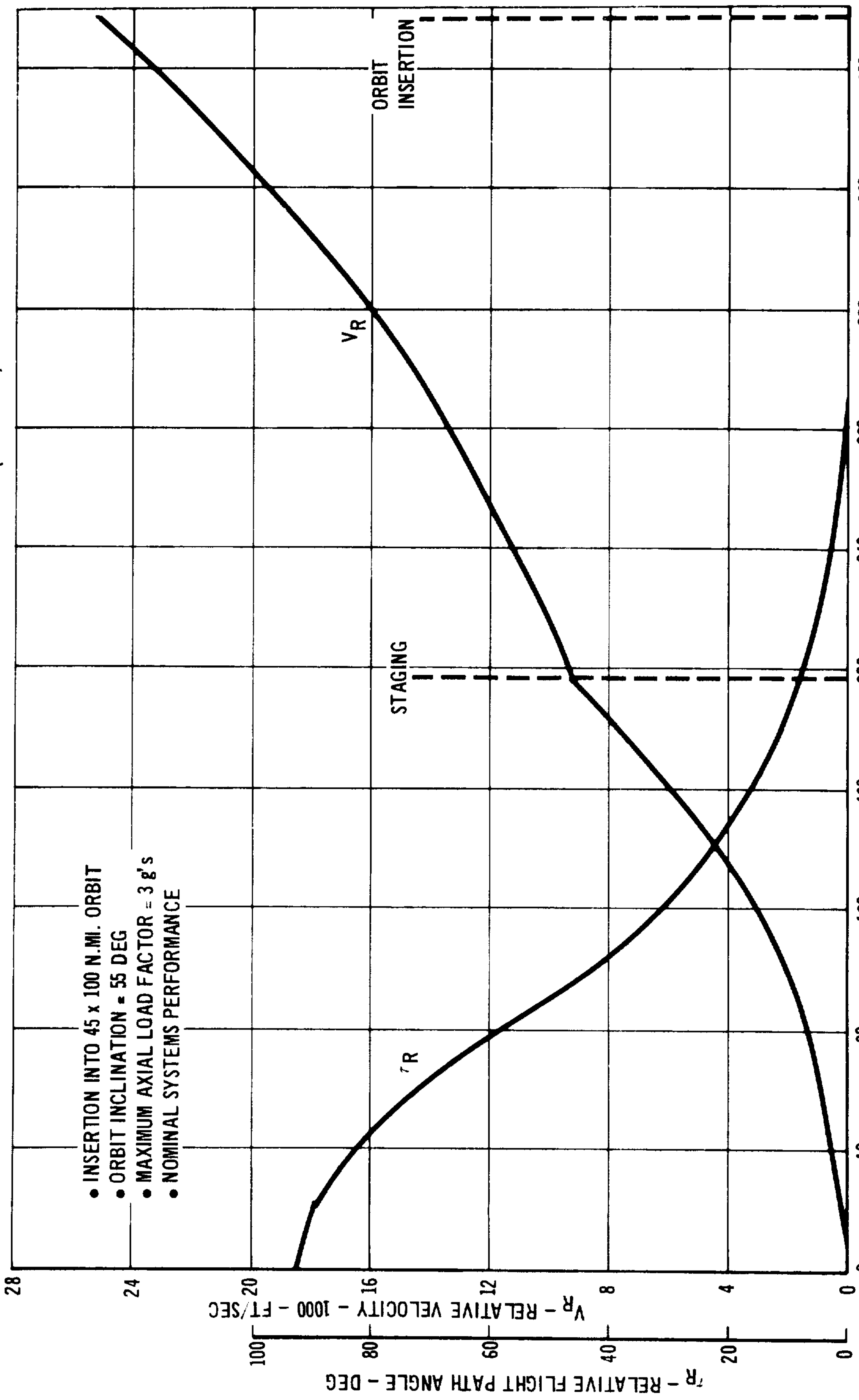


Figure 3-14

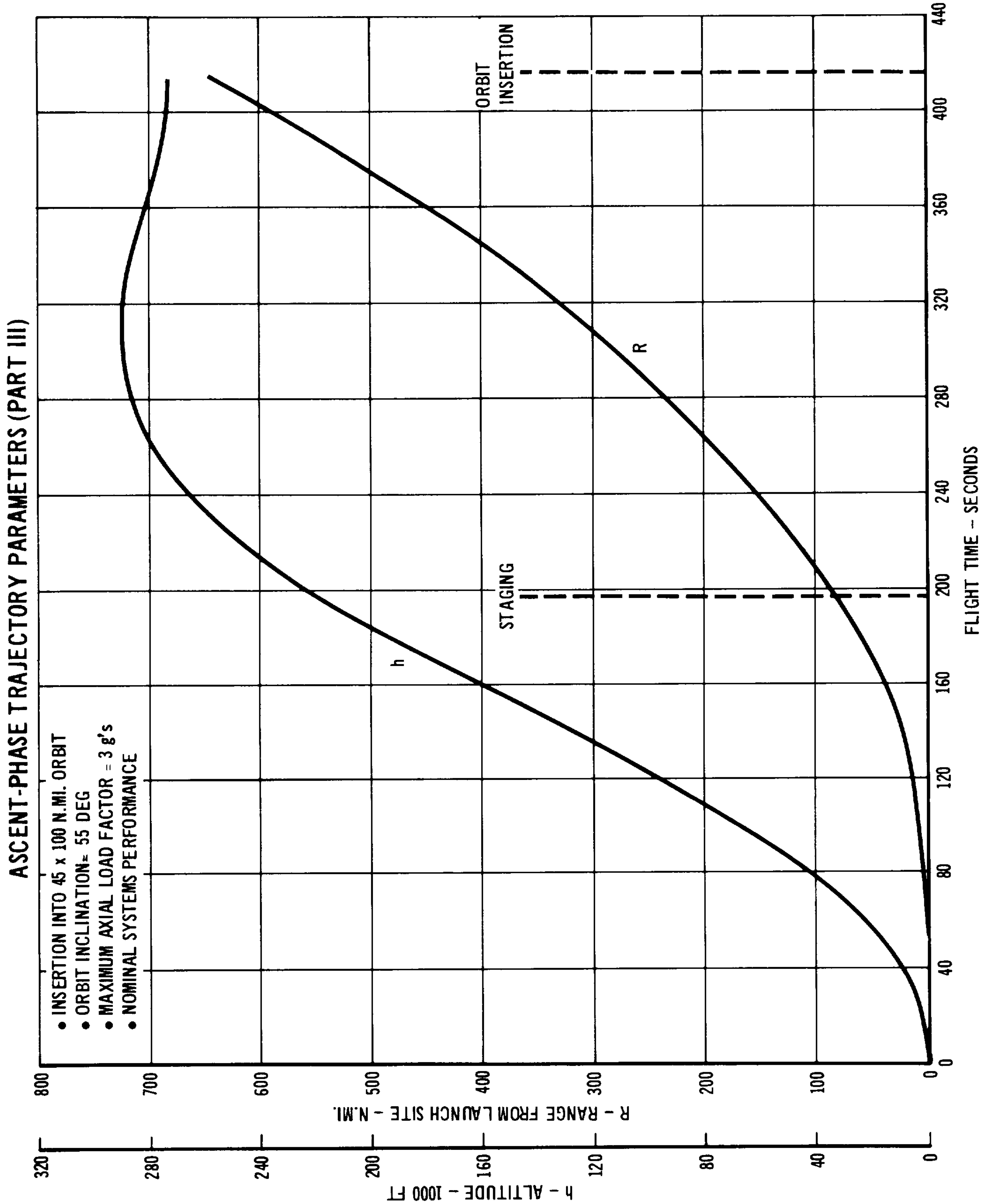

ASCENT-PHASE TRAJECTORY PARAMETERS (PART III)
• INSERTION INTO 45 x 100 N.MI. ORBIT
• ORBIT INCLINATION= 55 DEG
• MAXIMUM AXIAL LOAD FACTOR = 3 g's
• NOMINAL SYSTEMS PERFORMANCE
ORBIT INSERTION
STAGING
h
R
FLIGHT TIME – SECONDS
0
40
80
120
160
200
240
280
320
360
400
440
R – RANGE FROM LAUNCH SITE – N.MI.
0
100
200
300
400
500
600
700
800
h – ALTITUDE – 1000 FT
0
40
80
120
160
200
240
280
320

Figure 3–15

3.1.3 Stage Separation Analysis - A digital computer simulation of the separation dynamics of the Orbiter and the Carrier was performed. This simulation describes the motions of the centers of gravity of the two vehicles and determines whether or not the surfaces of the two vehicles intersect, that is, if a collision occurs. The basic assumptions of this simulation are that aerodynamic effects are ignored and that perfect control is maintained.

The approach for separating the Orbiter and Carrier had as its primary objectives; minimal disturbance to the flight path of the Orbiter, and minimum delay in thrust initiation of the Orbiter main engines. A technique which accomplishes these objectives to a reasonable degree is to cleanly separate the two vehicles, move the carrier away from the Orbiter and then fire the Orbiter main engines. Implementation of this technique is achieved through the following sequence of events.

a. The body rates and flight path rate are nulled.
b. The carrier engines are shut down. 4 seconds are required for the thrust to decay to zero.
c. After the carrier thrust is zero, a delay of .1 second elapses before separation occurs.
d. The attachment between the two vehicles is removed without disturbing either vehicle.
e. At the time of separation, a thrust of 16,000 pounds is applied by four 4,000 pound thrust engines, two forward of the carrier center of gravity and two aft. The center of thrust of the engines is 3 ft forward of the carrier center of gravity.
f. 1 second after separation, the Orbiter engines are ignited. Full thrust is obtained in 4.55 seconds.
g. 2 seconds after separation, the carrier separation thrust is terminated.

During this time the Orbiter body rates are maintained at null.

The separation trajectory using the above technique is shown in Figure 3-16. The orbiter translates across the path of the carrier. However, the carrier has rotated away from the orbiter and the orbiter has moved sufficiently forward when it crosses the flight path so that no collision occurs. The minimum clearance occurs in the first few seconds and the surfaces clear in this time period as shown in Figure 3-17.

SEPARATION TRAJECTORY

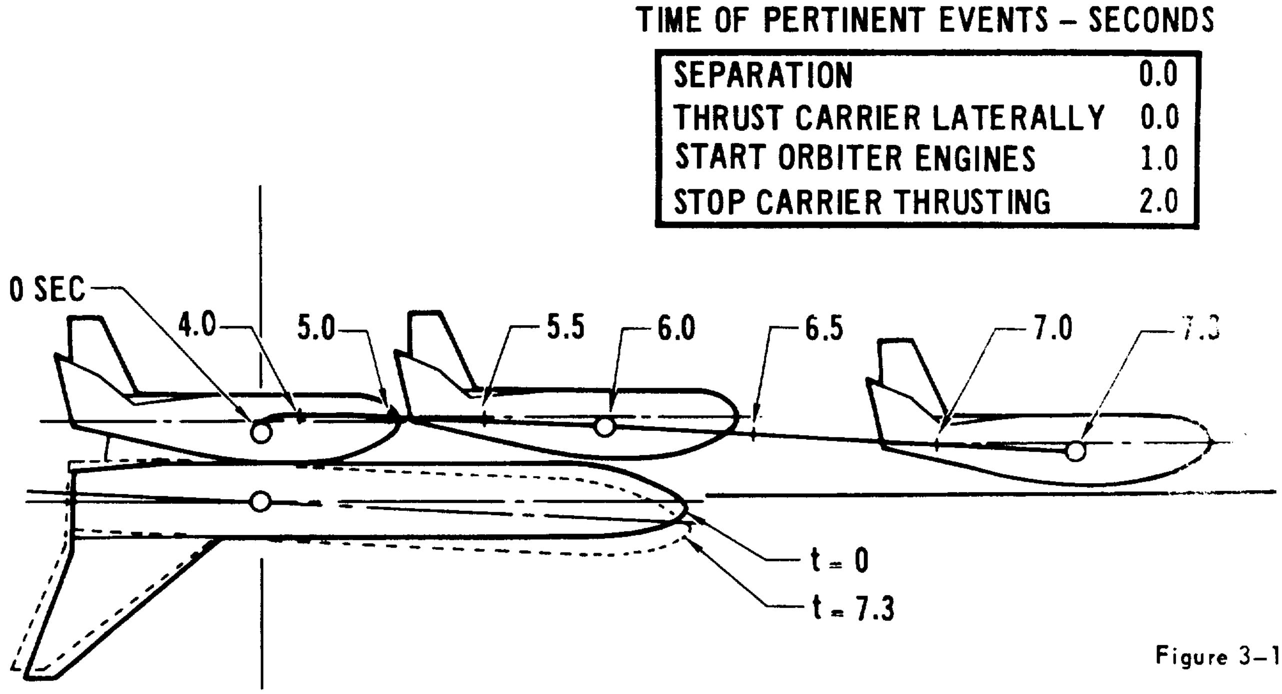


Figure 3–16

MOLD LINE SEPARATION

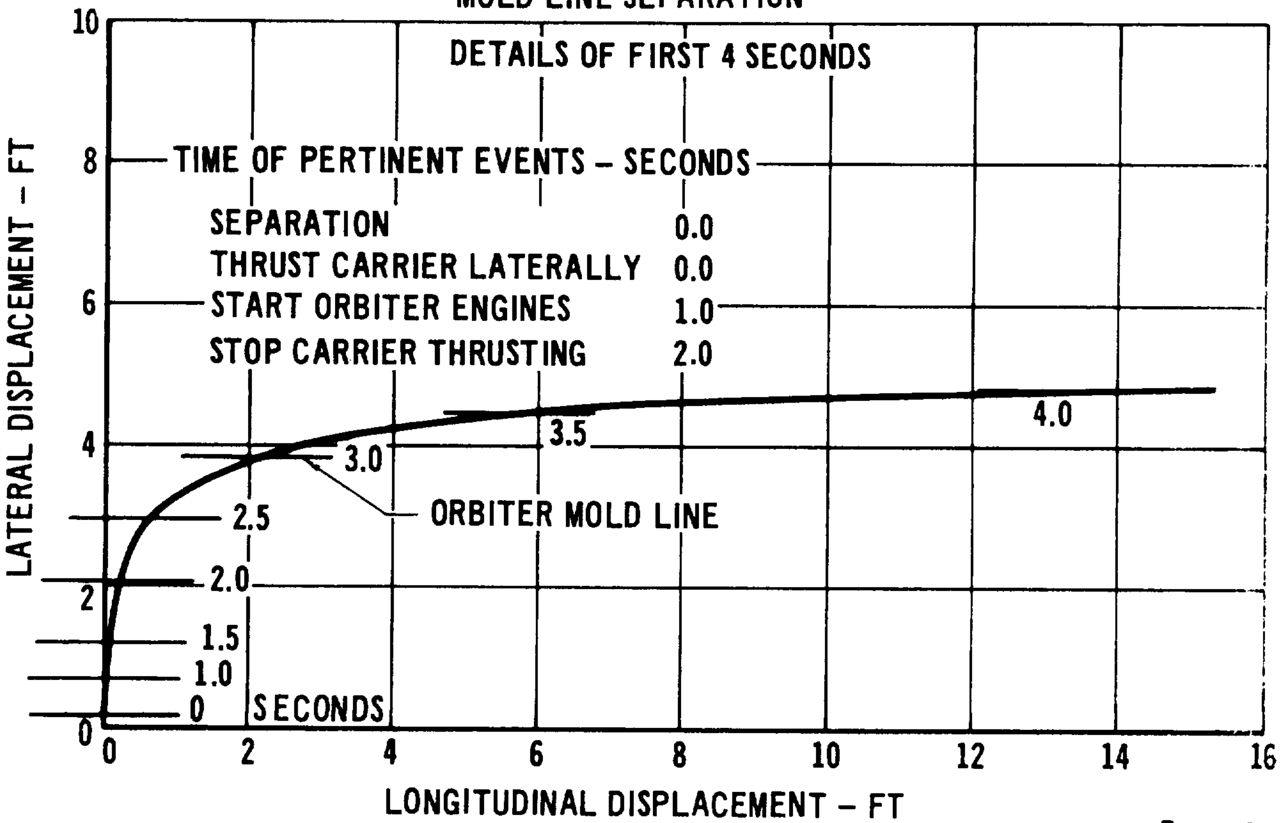


Figure 3–17

While this analysis did not consider dispersions, certain refinements were examined. For any forward positioning of the two forward separation engines, no collision occurs. A larger pitch motion of the carrier occurs which has the desirable effect of increasing the distance between the Orbiter and the carrier when the Orbiter passes in front of the carrier.

The additional equipment required to successfully perform a separation over that required for other orbital uses are the two extra thrusters forward of the center of gravity and a door to cover them during reentry.

3.1.4 Launch Heating Analysis - The nominal Carrier launch and reentry trajectories are shown in Figure 3-18. This figure shows that maximum stagnation point heating rates are more severe during reentry than launch. This applies to the entire Carrier except for the dorsal fin which experiences maximum heating during launch. Carrier temperatures are presented in Section 3.3.1-C of this volume.

Launch temperatures on the Orbiter for an impulsive velocity of approximately 14,500 ft/sec were predicted using the methods presented in Section 3.3.2. The effect of shock interaction between the Carrier and Oribter was not included in this analysis. Increases in local heating on the orbiter due to shock interference could result in heating rates being considerably increased prior to staging. The magnitude of this increase should be determined by heat transfer testing on the Carrier/Orbiter launch configuration. Temperature histories during launch at 12.5% of vehicle length are shown in Figure 3-19 for the side, lower surface centerline, upper surface, and upper surface centerline. Maximum side, lower surface centerline, upper surface and upper surface centerline temperatures are 1125°F, 1020°F, 830°F and 725°F, respectively, and occur at the time of insertion into a 45 nautical mile orbit. At insertion the angle of attack is reduced to zero degrees, altitude continues to increase and velocity decreases as a result of drag effects. Lower surface temperatures will decrease, whereas side and upper surface temperatures could increase slightly prior to decreasing. Further tests are required to more thoroughly establish heating rates in the low angle of attack regime. The temperatures are radiation equilibrium values based on a surface emittance of 0.85 and to not include an uncertainty factor.

The maximum launch temperatures shown, except for the upper surface centerline, are lower than the temperatures experienced during the nominal once/day reentry as seen in Figure 3-79, (3.3.2,C) and thus do not affect structural design. The maximum upper surface centerline launch temperature of 725°F is higher than the 680°F experienced during the nominal once/day reentry but remains below the maximum allowable temperature limit of titanium (1000°F).

CARRIER ASCENT AND REENTRY THERMAL ENVIRONMENT

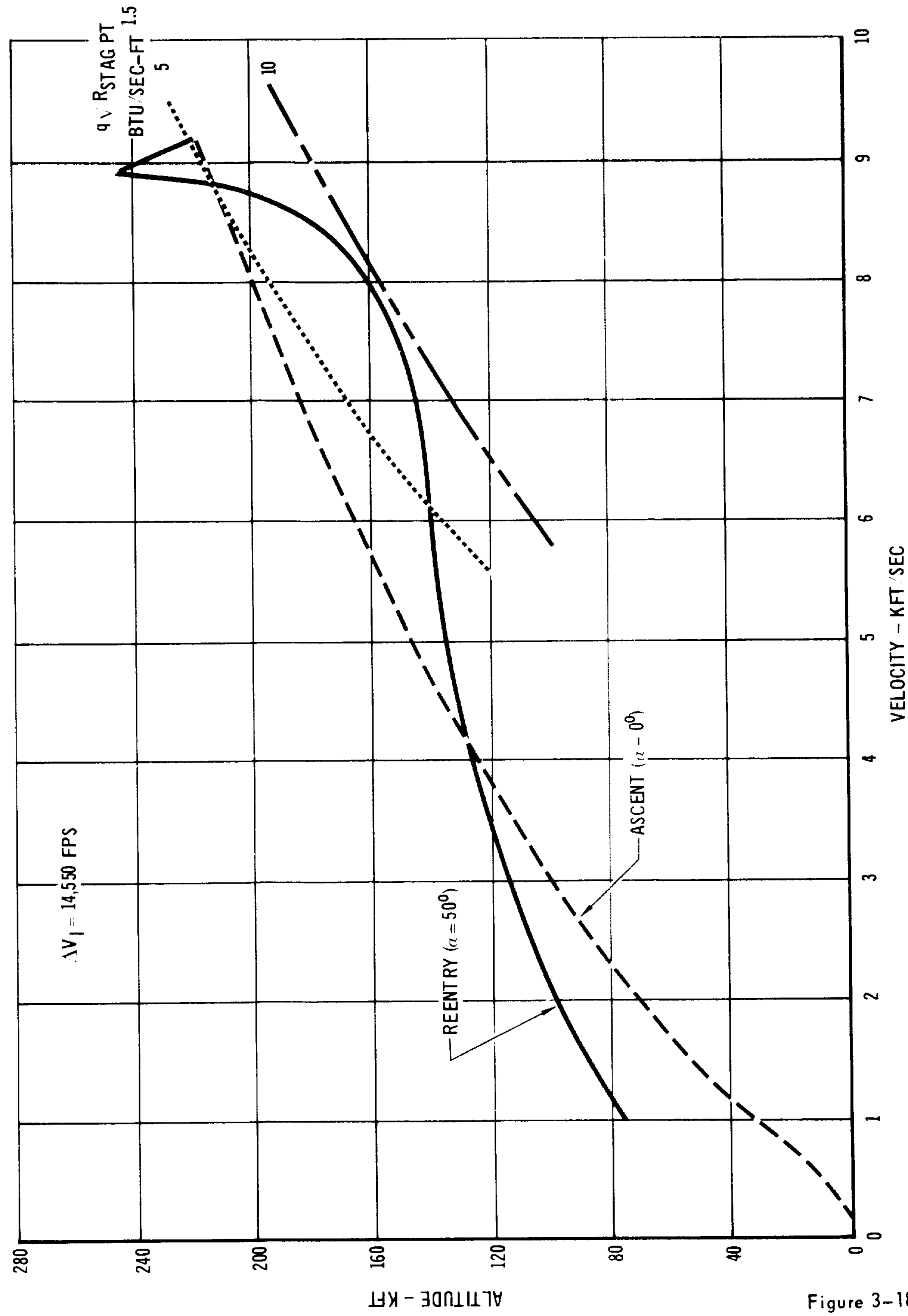


Figure 3-18

ORBITER TEMPERATURES DURING LAUNCH
(12.5% OF VEHICLE LENGTH)

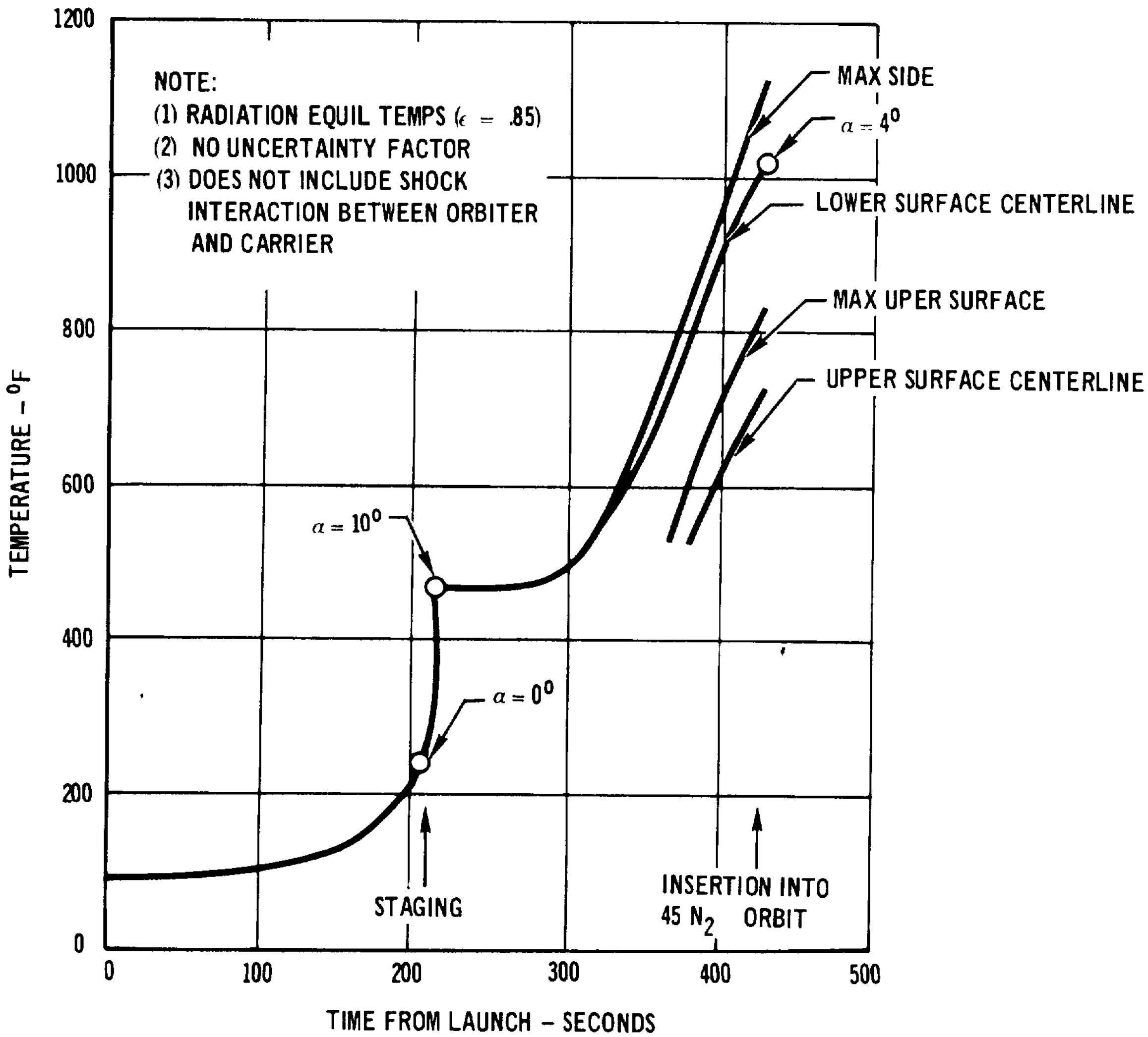


Figure 3-19

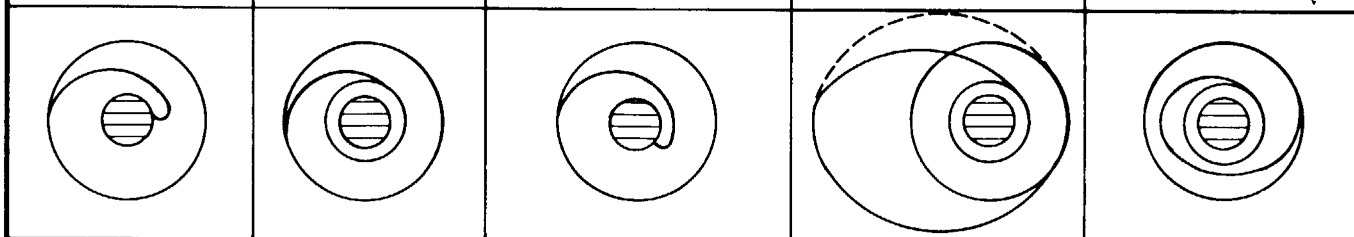
3.2 Orbital Performance

3.2.1 Gross Rendezvous Techniques - Five gross rendezvous techniques were evaluated for applicability to the near earth logistic mission. The advantages and disadvantages of each technique are summarized in Figure 3-20. In the ground-hold-phasing technique, the launch is delayed until the phasing with the space station is such that injection would occur at the perigee of the transfer orbit. For this technique, the ΔV for the inherent plane change is very large but the transfer time is a minimum. For the parking orbit phasing technique, the ΔV is a minimum but the time from launch through docking exceeds the established maximum for much of the range of space station altitudes and inclinations. The rendezvous compatible orbit technique is a restricted form of the ground hold phasing technique wherein the space station ground track repeats in a reasonable number of orbits but the restrictions are too great to be considered for a generalized space station. The limited rev technique is a highly flexible technique which provides a trade-off between ΔV and time in orbit. Sufficient plane change capability is provided to allow for moderate ground hold phasing. The remainder of the phasing is accomplished in a phasing orbit. Both catchup and drop-back oribts are used to limit the time in orbit. A modified version of the limited rev technique is recommended for the baseline mission. In the modified limited rev technique, the drop-back maneuver, which requires a sizeable ΔV, is eliminated at the cost of increasing the phasing time. In either of the limited rev techniques, the plane change ΔV is vector summed with the three bielliptic transfer maneuvers to reduce the total ΔV requirement.

3.2.2 Gross Rendezvous Velocity and Phasing Requirements - The transfer velocity requirement to enable a logistic spacecraft to leave a circular 100 n.m. earth orbit to rendezvous with a space station at various altitudes with a capability to provide from zero to one degree plane change is given in Figure 3-21. The added plane change ΔV is based on performing the maneuvers as explained for the modified limited rev technique. For the baseline mission (55 degree inclination, 270 n.m. orbit), a 540 ft/sec ΔV is required with zero plane change capability. To reach a 300 n.m. orbit with one degree plane change capability, 775 ft/sec is required.

To achieve a rendezvous operation within the prescribed 24 hour total ascent time constraint, the maximum parking orbit phasing time should be limited to 20 hours. Also assuming worst-case phasing and a launch-any-day capability, the plane change required will be dependent on meeting these constraints. In addition,

NEAR-EARTH ORBIT GROSS RENDEZVOUS TECHNIQUES



GROUND HOLD PHASING	PARKING ORBIT PHASING	RENDEZVOUS COMPATIBLE ORBITS	LIMITED REV	MODIFIED LIMITED REV √
ADVANTAGES				
1. MINIMUM TIME IN ORBIT	1. MINIMUM ΔV	1. MINIMUM TIME IN ORBIT 2. MINIMUM ΔV	1. MOST FLEXIBLE (TRADE ΔV FOR TIME)	1. FLEXIBLE 2. LOW ΔV 3. LOW TIME IN ORBIT AT LOW INCLINATIONS
DISADVANTAGES				
1. LARGEST ΔV FOR HIGH INCLINATIONS	1. LONGEST TIME IN ORBIT 2. MUST BE COPLANAR	1. STATION ORBIT MUST BE CLOSELY CONTROLLED 2. STATION AND SPACECRAFT LAUNCH TRAJECTORIES MUST BE COMPATIBLE	1. DROP BACK COSTS ΔV 2. FAIRLY LONG TIME IN ORBIT FOR HIGH INCLINATIONS	1. ELIMINATING DROP BACK INCREASES TIME 2. LONG TIME IN ORBIT FOR HIGH INCLINATIONS

√ RECOMMENDED

Figure 3–20

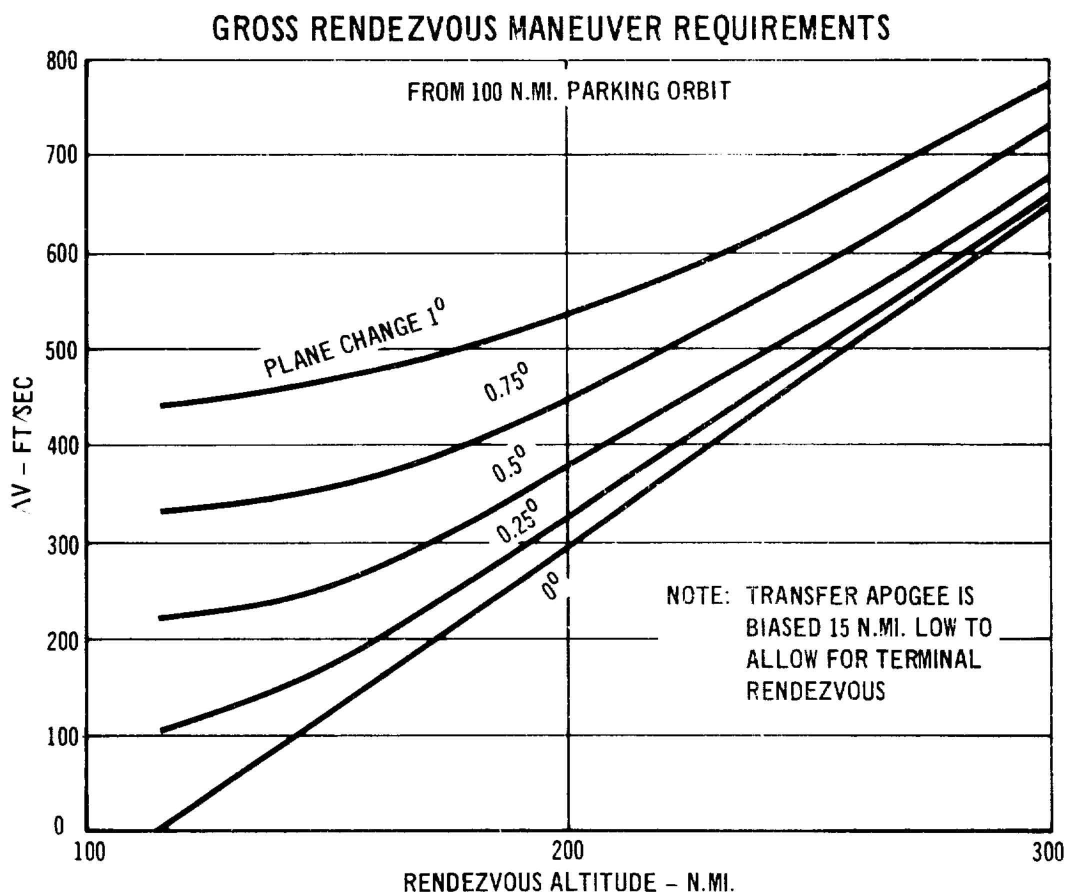

GROSS RENDEZVOUS MANEUVER REQUIREMENTS
FROM 100 N.MI. PARKING ORBIT
ΔV – FT/SEC
0
100
200
300
400
500
600
700
800
RENDEZVOUS ALTITUDE – N.MI.
100
200
300
PLANE CHANGE 1°
0.75°
0.5°
0.25°
0°
NOTE: TRANSFER APOGEE IS BIASED 15 N.MI. LOW TO ALLOW FOR TERMINAL RENDEZVOUS

Figure 3-21

the current ETR launch constraints call for launch headings between 44 and 110 degrees. The southern and northern boundary limitations are exceeded for orbital inclinations greater than 34 and 52 degrees, respectively. These constraints are not compatible with in-plane launches into the nominal 55 degree inclination orbit without boost yaw steering. Hence, modified launch azimuth constraints of 35 to 180 degrees were assumed for this study. With these modifications, the launch heading reaches the northern boundary at an inclination of 60 degrees and the southern boundary at an inclination of 90 degrees.

The maximum orbit phasing time for an in-plane (zero plane change allowance) Hohmann transfer from a 100 n.m. parking orbit is shown as a function of space station altitude in Figure 3-22 for orbit inclinations of 55 degrees and 90 degrees. Also shown are the decreased phasing times achievable with a one degree plane change allowance. Note that the 24 hour ascent requirement can be met for the baseline mission without a plane change.

3.2.3 Launch Opportunities - The effect of restricting the transfer ΔV to 775 ft/sec upon launch opportunity is illustrated in Figure 3-23. A launch may not be possible every day for the lower station altitude with orbit inclinations greater than 60 degrees. Figure 3-24 shows that 1900 ft/sec transfer ΔV is required to provide a once-a-day launch opportunity for missions defined by the shaded region of Figure 3-24. However, other schemes such as yaw steering during boost would be preferable to this large transfer ΔV requirement.

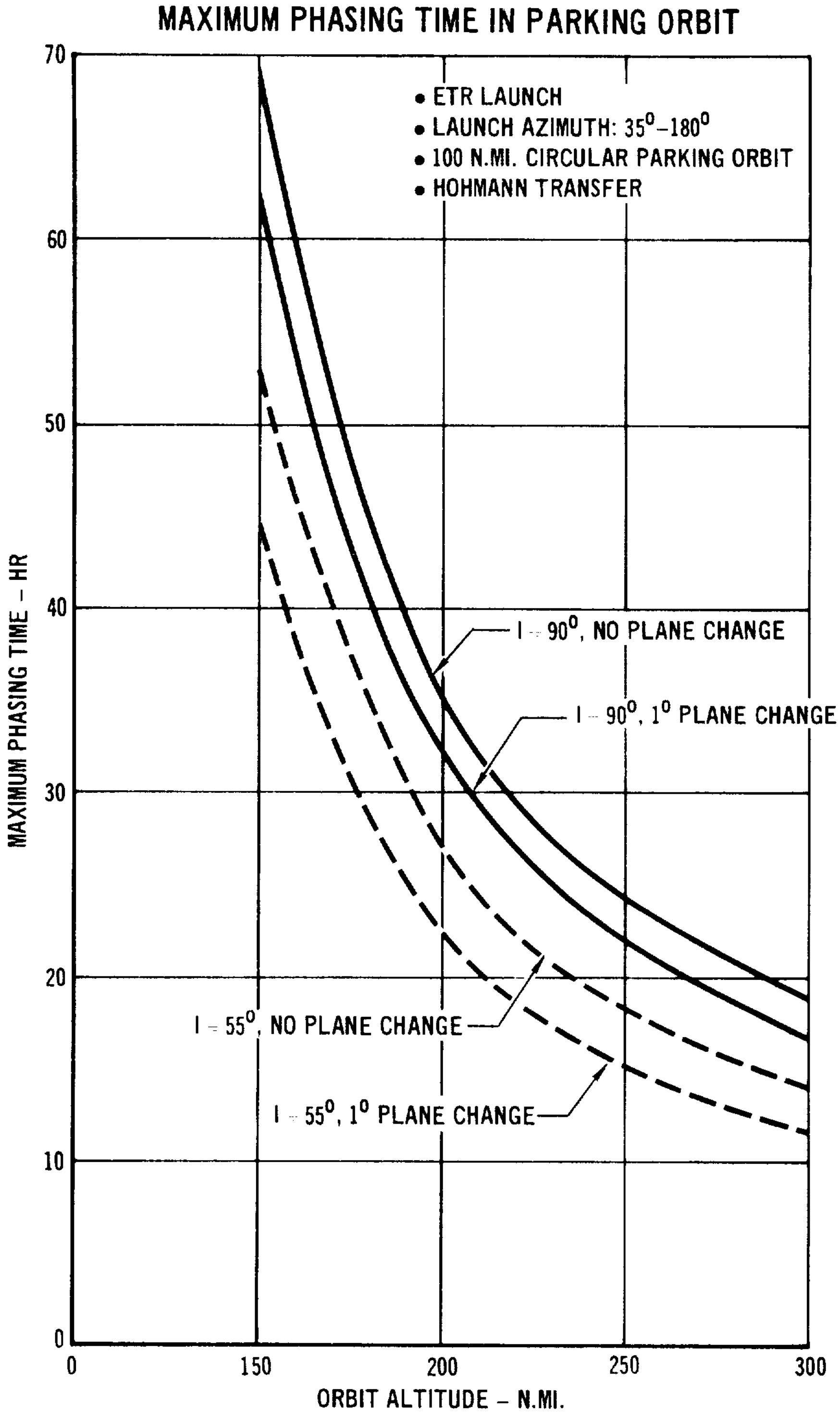

MAXIMUM PHASING TIME IN PARKING ORBIT
• ETR LAUNCH
• LAUNCH AZIMUTH: 35°–180°
• 100 N.MI. CIRCULAR PARKING ORBIT
• HOHMANN TRANSFER
I = 90°, NO PLANE CHANGE
I = 90°, 1° PLANE CHANGE
I = 55°, NO PLANE CHANGE
I = 55°, 1° PLANE CHANGE
MAXIMUM PHASING TIME – HR
ORBIT ALTITUDE – N.MI.
0
10
20
30
40
50
60
70
150
200
250
300

Figure 3–22

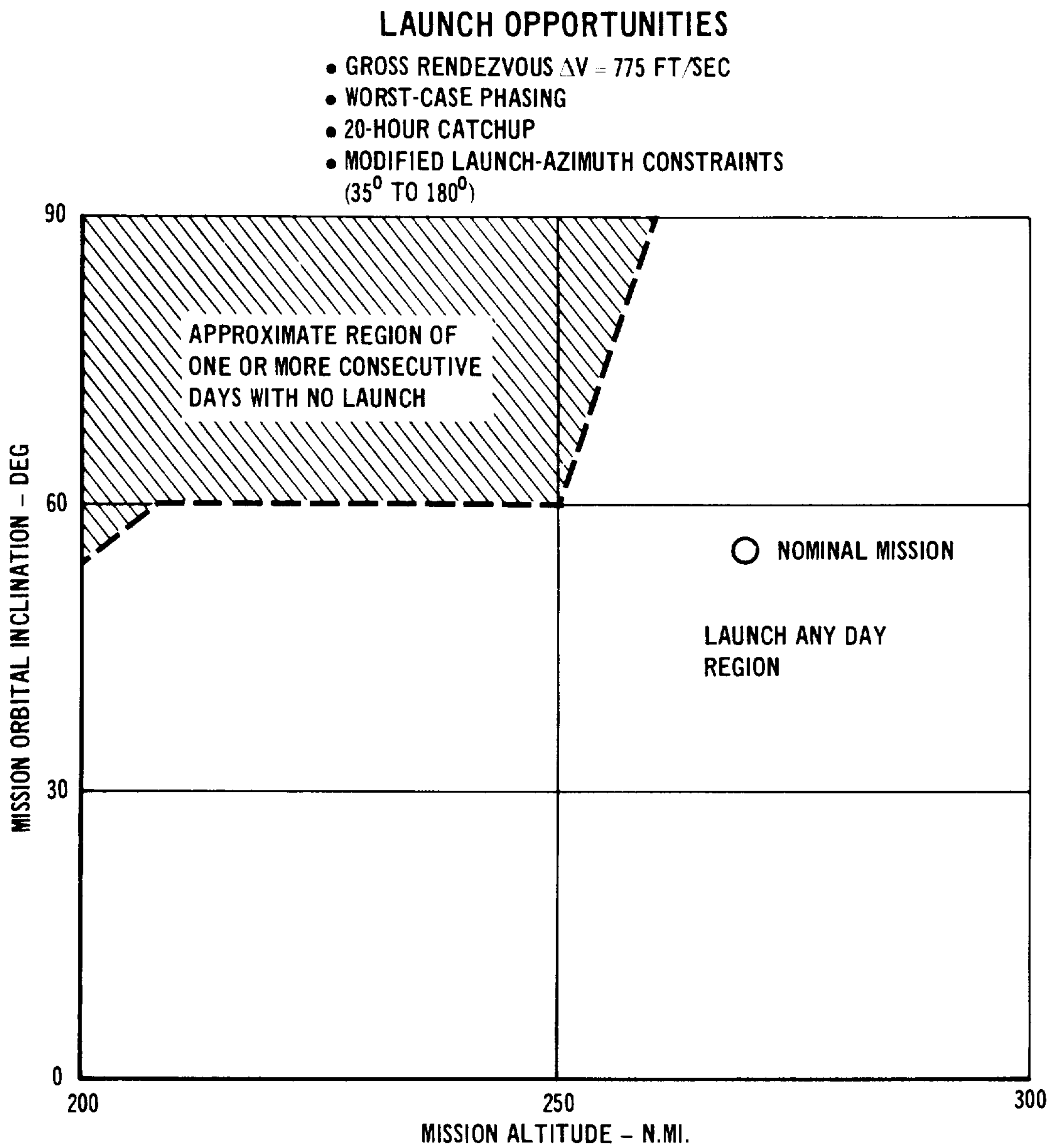

LAUNCH OPPORTUNITIES
• GROSS RENDEZVOUS ΔV = 775 FT/SEC
• WORST-CASE PHASING
• 20-HOUR CATCHUP
• MODIFIED LAUNCH-AZIMUTH CONSTRAINTS (35° TO 180°)
APPROXIMATE REGION OF ONE OR MORE CONSECUTIVE DAYS WITH NO LAUNCH
NOMINAL MISSION
LAUNCH ANY DAY REGION
MISSION ORBITAL INCLINATION – DEG
90
60
30
0
200
250
300
MISSION ALTITUDE – N.MI.

Figure 3–23

MAXIMUM PHASING TIME IN PARKING ORBIT VS ΔV

100 N.MI. PARKING ORBIT
SPACE STATION ALTITUDE – 200 N.MI.
90^{0} ORBIT INCLINATION

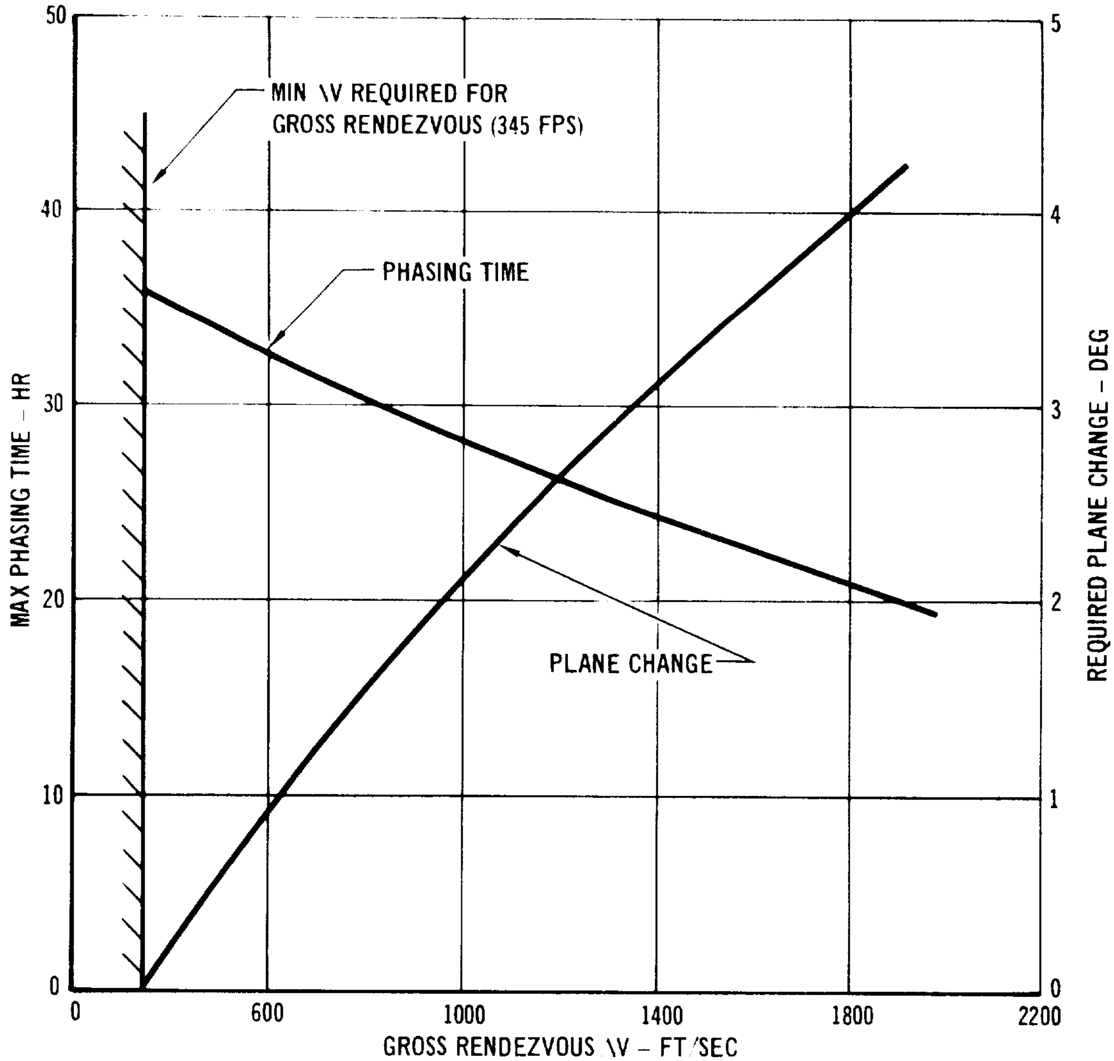


Figure 3–24

3.2.4 Orbital Maneuver Velocity Requirements - For the baseline mission, with the space station in a 270 nautical mile circular orbit at an inclination of 55 degrees, the vehicles are launched in plane. Abort considerations and range constraints dictate whether a northerly or southerly launch is used. The orbiter is injected into a 45 x 100 nautical mile orbit. At the first apogee the orbit is circularized at 100 nautical miles. At this altitude the orbiter catches up with the space station. If the orbiter must remain at this altitude for 24 hours, a 40 feet per second drag make-up is required. If errors in the inclination of the 100 nautical mile parking orbit occurred, a plane change to correct the inclination would be incorporated in the orbit transfer burns. For an error of .2 degrees, an additional 10 feet per second is required. The orbit transfer is a Hohmann type with the final altitude being 255 nautical miles.

The final altitude is selected to be 15 nautical miles below the space station with the final transfer burn occurring in a trailing position from the space station. From this slow catchup orbit, terminal rendezvous is initiated when the trailing displacement is about 60 nautical miles. An incremental velocity of 60 feet/second will perform the terminal rendezvous but the results of Gemini flight experience indicate that a 150 to 400 percent increase is required to perform these maneuvers. At completion of the terminal rendezvous, the orbiter and space station are in close proximity with the relative rates being nulled. From this condition, the orbiter performs the necessary station keeping maneuvers. For a seven day on-orbit mission, the relative position of the orbiter and space station will require trimming of the orbiter orbit. Prior to retrograde, a return phasing maneuver may be required to provide for a once a day return capability. For a landing site at 28.5 degree latitude and an orbiter inclination of 55 degrees, about a 400 nautical mile cross range capability is required. While the orbiter has this much aerodynamic cross range capability other design criteria may limit the use of the full potential. Assuming no cross range is utilized, a 285 ft/second phasing maneuver would be required for the worst case. Without return orbit phasing the incremental velocity is 400 feet/second for a 1 degree entry angle, and with the worst case phasing this could be 500 to 760 feet/second, depending on whether retrograde is performed at apogee or perigee. It is always possible to orient the phasing orbit so that retrograde can be at apogee. For a 1.5 degree entry angle, a deorbit impulse of 425 feet/second is required for the baseline.

A summary of the incremental velocity requirements is shown in Table 3-4. A MDAC estimate is included with the NASA specified. The estimate for the gross rendezvous varies from that required to perform the nominal mission (270 n.m., 55 degree inclination, no plane change capability) to that required for a 300 n.m, 90° inclination with 1 degree palne change capability. This maximum value is not sufficient to cover the full launch opportunity spectrum with worst case phasing as shown in Figures 3-23 and 3-24. However, it is felt that the majority of the missions can be handled, and if worst case phasing is not encountered, the 24 hour ascent time requirement may be accomplished.

Table 3-4

POST-INJECTION INCREMENTAL VELOCITY REQUIREMENTS
(Ft/Sec)

Mission Phase	MDAC Estimate	NASA Specified
Circularization	100	100
Launch Dispersions		200
Drag Makeup	40	
Orbit Transfer	540-775*	558
Plane Change	10	
Terminal Rendezvous	200	142
Station Keeping	40	
Retrograde	425-470**	500
Subtotal	1355-1635	1500
Contingency	135-163	500
Total	1490-1798	2000

* 540 ft/sec for baseline mission; 775 ft/sec provides capability to rendezvous for sttation altitudes between 200 and 300 NM with orbit inclinations between 28.5 and 90 deg. NOTE: The 24 hour ascent requirement is not satisfied at the lower altitudes for inclination greater than 60 deg.

** 425 ft/sec = retrograde ΔV for 270 NM orbit.
470 ft/sec = retrograde ΔV for 300 NM orbit.

3.3 Entry Performance - The Orbiter must enter the earth's atmosphere following retrograde from the nominal 270 NM altitude circular orbit. The Carrier, however, must return from a point slightly higher than the staging altitude but well within the sensible atmosphere. The aerodynamics, trajectories and heating analyses for both the Orbiter and Carrier associated with these entry requirements are presented and discussed in this section. The entry performance data for the Carrier and Orbiter vehicles are considered separately and in that order.

3.3.1 Carrier Performance - The Carrier entry performance analyses are presented in the following paragraphs. The vehicle, aerodynamic data are discussed first followed by the trajectory and heating considerations.

a) Carrier Entry Aerodynamic Analysis - During the study a series of exploratory wind tunnel tests of the Carrier were performed at the Langley Research Center. The tests included the Carrier alone as well as the 2-stage ascent phase configuration. Force and moment tests were conducted at a subsonic Mach number of 0.3 and a hypersonic Mach number of 10.4. Thermographic tests on the Carrier alone were conducted at Mach number 10.4, using phase-change material to indicate first order heating effects. The aerodynamic forces for both the Carrier and the ascent configuration have been normalized with the Carrier theoretical wing area and the moments were normalized with the wing area and the corresponding mean aerodynamic chord. In the figures which present longitudinal moments for the carrier alone, the moment reference point was positioned at the 66% station on the vehicle centerline. Ascent configuration data are presented for moment reference of 45 and 74.1 percent for subsonic and hypersonic data respectively. The total planform area of the Carrier is 26% greater than the theoretical wing area and the mean aerodynamic chord is 44% of the body length. A control deflection convention was adopted which defined negative deflection as trailing edge up. The sketch in Figure 3-25 presents the plan view of the Carrier wind tunnel configuration and defines the region used for the aerodynamic reference area. The Carrier model was an intermediate configuration and did not exactly match the final study configuration; however, the model lines were sufficiently close to insure that the trends of the test data are good representations of the vehicle characteristics.

CARRIER CONFIGURATION
WIND TUNNEL MODEL PLANFORM

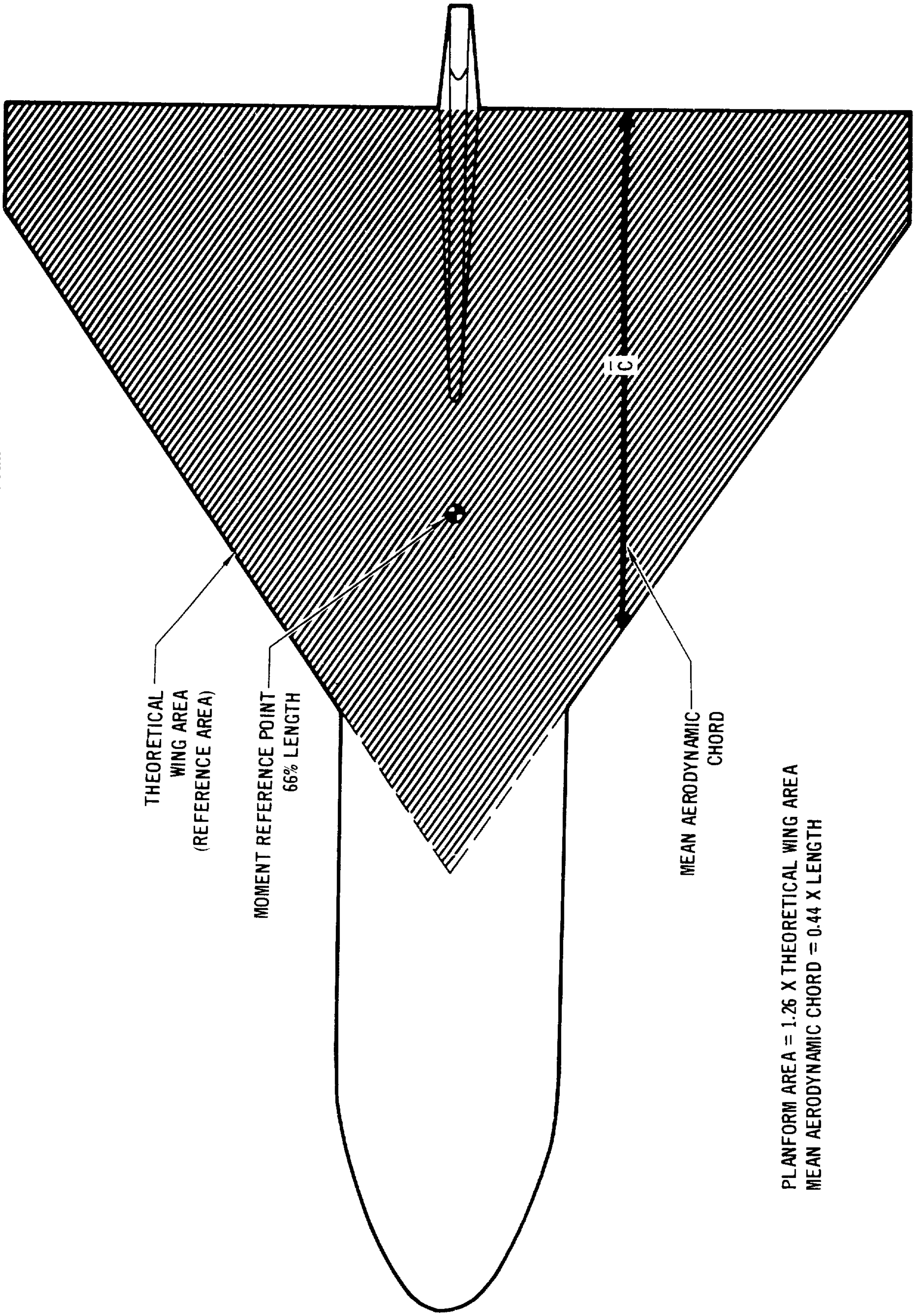


Figure 3–25

The summary table, shown on Figure 3-26 lists the range of pertinent test variables which were obtained at the two Mach number conditions.

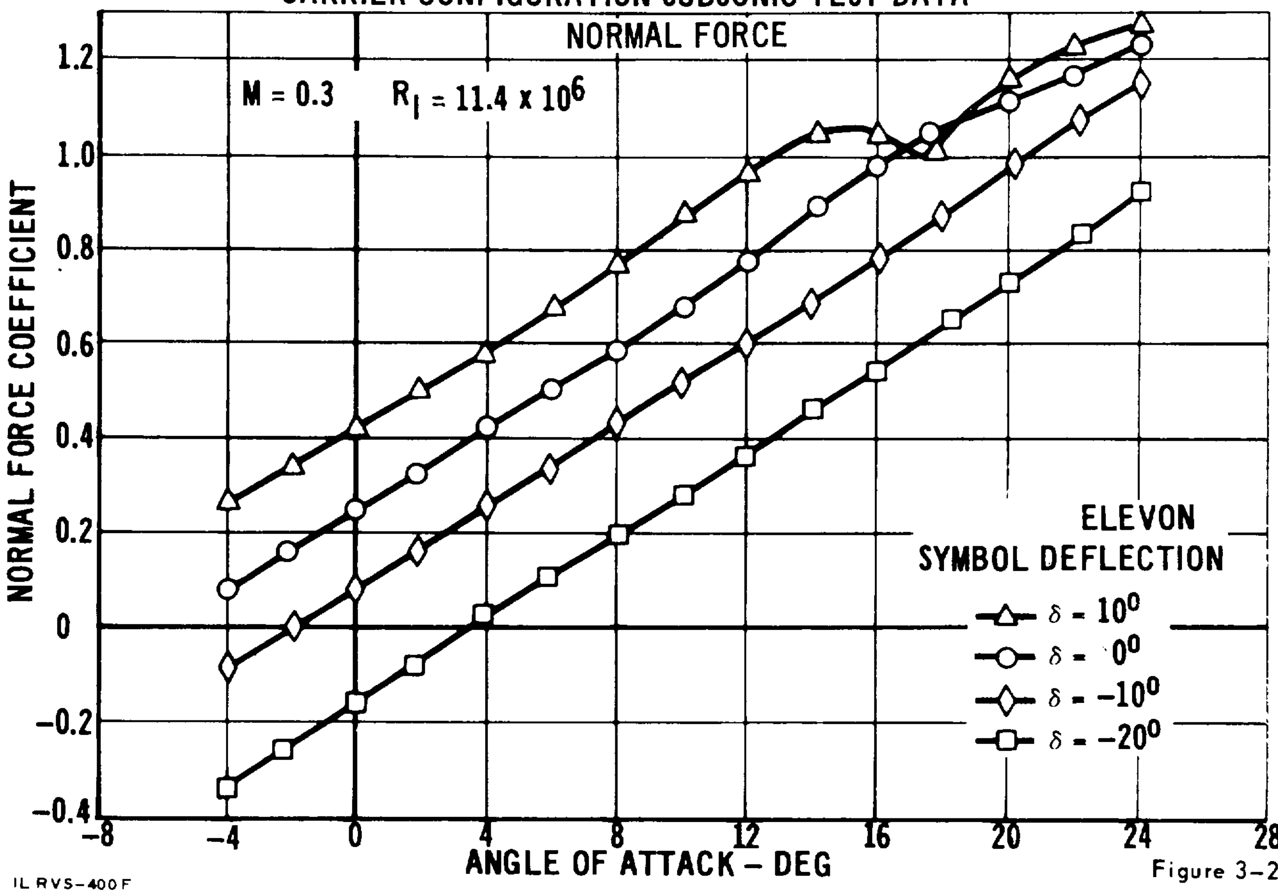
- Subsonic Aerodynamics - Figures 3-27 through 3-30 present the subsonic lift, drag, and moment characteristics of the Carrier configuration for four elevon deflections. The Carrier model wing was constructed with a NACA 4415 airfoil section, which provided linear normal force variations up to approximately 14° angle of attack. Beyond this point, the test data obtained with positive elevon deflection exhibited wing stall characteristics. However, the data with the negative deflections, which are presently being used for trim, did not show any stall characteristics over the angle-of-attack range included in the test. These data yield a zero deflection-zero angle-of-attack normal force value of 0.25 and a normal force slope of 0.044 per degree. The normal force variation with control deflection is nonlinear, but for the negative deflections the value tends to be of the order .023 per degree. This value of elevon effectiveness is due mainly to the size of the elevons, since the combined plan area of both panels is approximately 18% of the theoretical wing area. The pitching moment figure defines the stability characteristics of the Carrier configuration. As the figure indicates, the vehicle is stable over the entire trim angle-of-attack range shown, but exhibits undesirable moment characteristics in the untrimmed region where wing stall occurs. The configuration has a static margin of approximately 1%, thus a moment reference point at 67% would cause the vehicle to be neutrally stable at moderate angles of attack. Although the zero deflection configuration has a large residual pitching moment (-0.089), the elevon control effectiveness is sufficient to offset this value and provide for trim up to at least 24° angle of attack. The figure which presents the untrimmed Lift-Drag ratios shows that the maximum L/D is obtained with negative elevon deflection. The maximum value of L/D is 7.65 and is obtained with a negative 10° deflection at 7° angle of attack. A cross-plot of the test data indicates that the maximum obtainable value of L/D is 7.72; however, this is an ideal value, since the vehicle would have to trim in an unstable condition to utilize the angle of attack and control deflection necessary to obtain this maximum. The subsonic trim figure shows that the

CARRIER TEST SUMMARY

LRC LOW TURBULENCE PRESSURE TUNNEL (M = 0.30)	LRC CONTINUOUS FLOW HYPERSONIC TUNNEL (M = 10.4)
FORCE TESTS • CARRIER $-7^0 < \alpha < 24^0$ $-20^0 < \delta_e < +10^0$ $-5^0 < \beta < +0^0$ $\frac{t}{c} = 0.09, 0.15$ HIGH AND LOW WING WING FAIRING OFF AND ON • ASCENT CONFIGURATION $-7^0 < \alpha < 24^0$ $-5^0 < \beta < 5^0$	FORCE TESTS • CARRIER $-2^0 < \alpha < 60^0$ ELEVON ON AND OFF • ASCENT CONFIGURATION $-4^0 < \alpha < 10^0$ HEAT TRANSFER • CARRIER M = 10.4 $\alpha = 15^0$ AND 50^0 HIGH AND LOW WING FAIRING OFF AND ON • MATED CONFIGURATION M = 10.4

Figure 3-26

CARRIER CONFIGURATION SUBSONIC TEST DATA
NORMAL FORCE



IL RVS-400F

Figure 3-27

CARRIER CONFIGURATION SUBSONIC TEST DATA
STABILITY CHARACTERISTICS

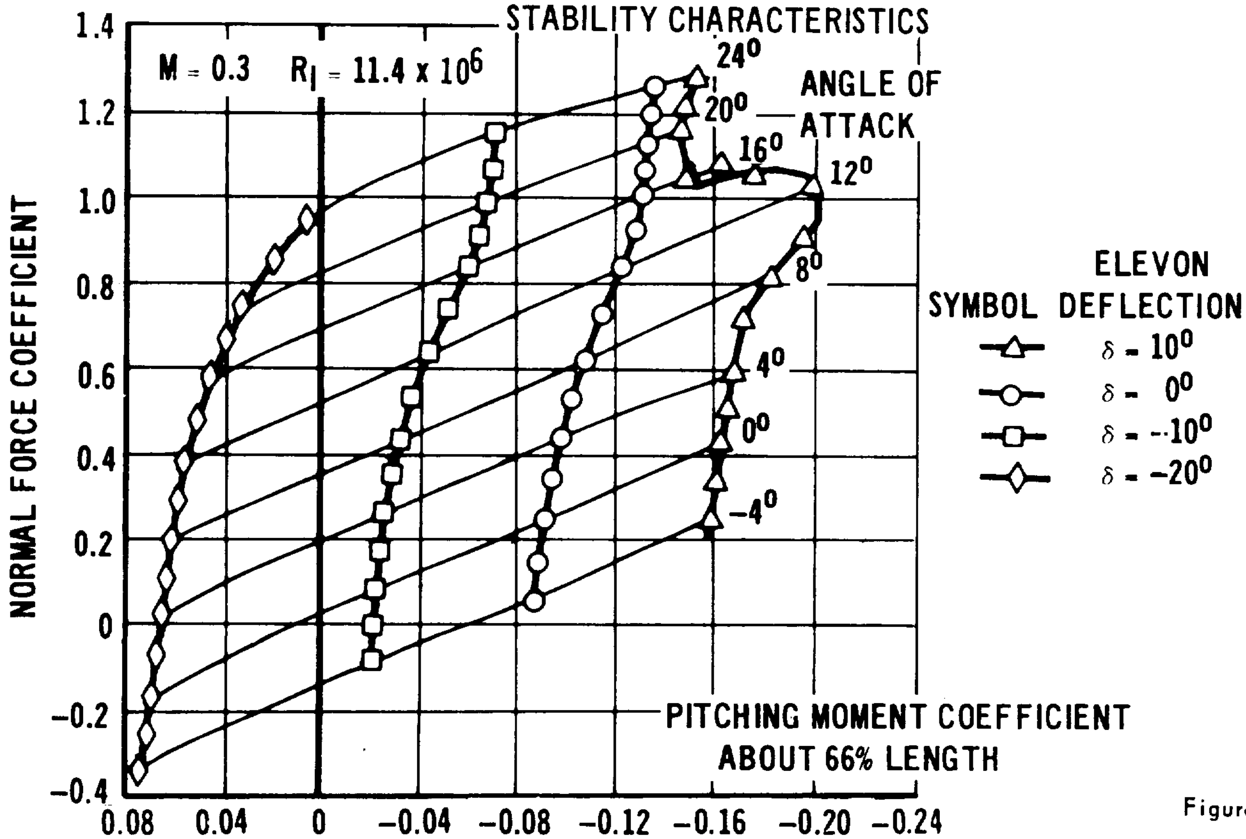
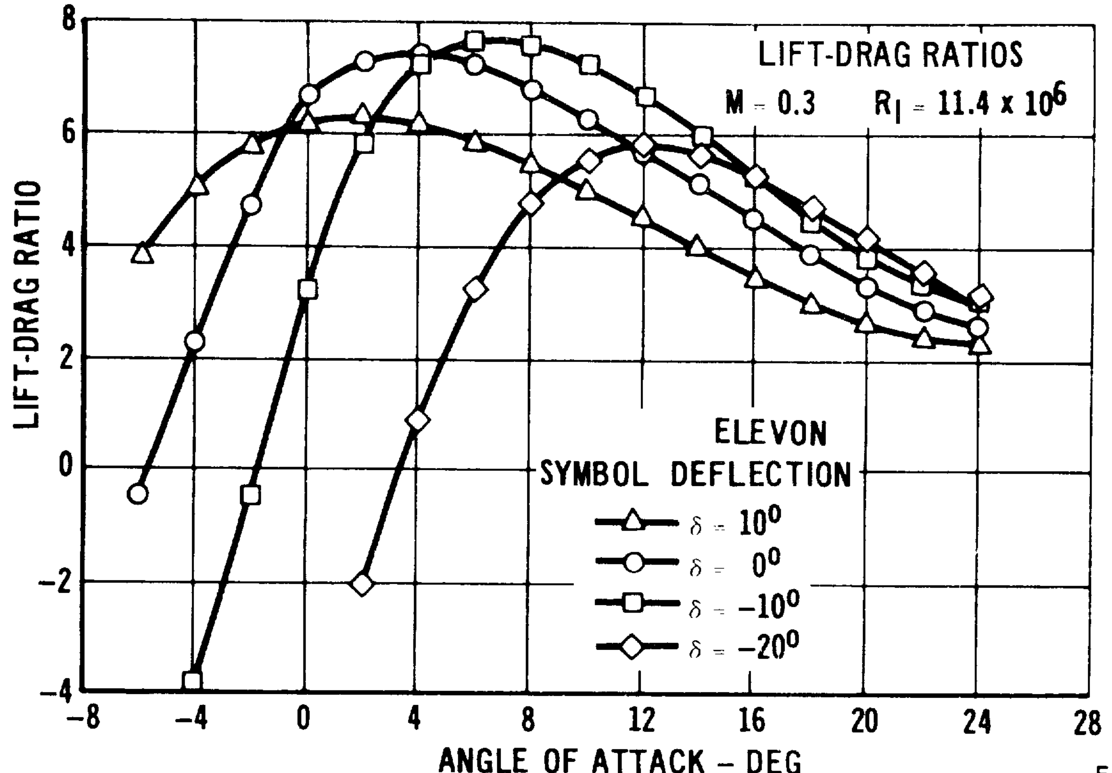


Figure 3-28

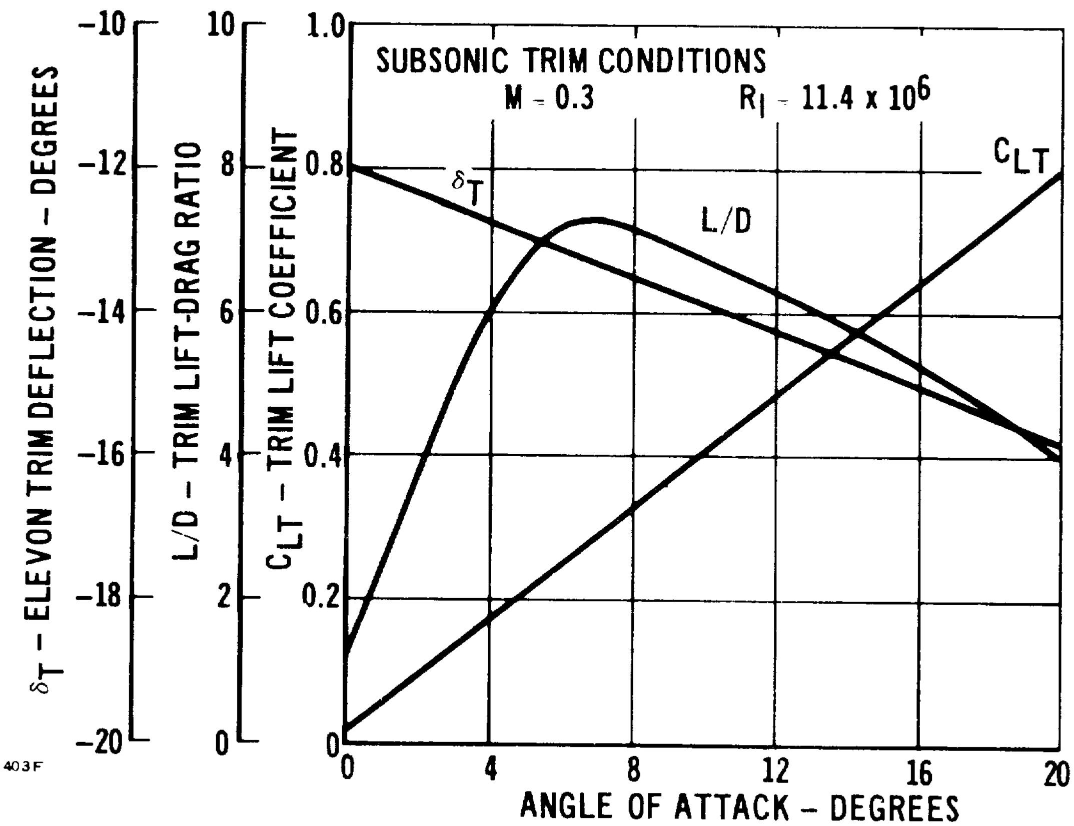
CARRIER CONFIGURATION SUBSONIC TEST DATA



ILRVS-402F

Figure 3-29

CARRIER CONFIGURATION



ILRVS-403F

Figure 3-30

maximum trim L/D is 7.4, which occurs at 7.5° angle of attack with a negative 13.4° elevon deflection. The vehicle has a sufficient amount of trim control authority, since one degree of control deflection can increment the trim angle of attack by 5.5°. The trim characteristics were obtained by working directly with the test data, and at present no corrections have been made to account for full-scale effects.

o <u>Subsonic Configuration Sensitivities</u> - During the exploratory wind tunnel tests, several Carrier wing geometry variations were introduced into the run schedule. These included a wing-tail as well as the clipped delta configurations. The baseline configuration is a clipped delta utilizing a 15% thick, low wing, without a leading edge fairing. The wing geometry variations for the clipped delta included a change in the vertical location, a change in the wing thickness, and the addition of a leading edge root fairing.

1) <u>Wing-Tail Configuration</u> - In order to obtain greater insight regarding the optimum carrier configuration, it was decided to analyze: (1) a tailless clipped delta, because of its small center of pressure travel with Mach number and (2) a wing tail because of its better subsonic L/D. The following constraints were placed on the carrier design: (1) both vehicles would have similar body shape and length; (2) the combined theoretical planform areas of the wing and tail would not exceed the theoretical clipped delta planform area.

Initial subsonic and hypersonic stability estimates of the wing-tail shape indicated a large longitudinal instability for the straight wing corresponding to a c.g. located 66 percent of the body length aft of the nose, requiring both wing sweep and an increase in tail size to produce a statically stable vehicle. The required rearward wing location and the large amount of sweep resulted in only a small distance between the wing trailing edge and the tail leading edge such that for all practical purposes the wing-tail had evolved to a delta. The clipped delta was thus selected as the baseline carrier configuration for the midterm presentation. It was decided, however, to obtain wind tunnel data for both configurations to verify the analytical results. Static longitudinal aerodynamic characteristics obtained at Mach 0.3 at a Reynolds number of 11.4 million in the LRC Low Turbulence Pressure Tunnel are shown in Figures 3-31, 3-32 and 3-33. These results indicate neutral longitudinal stability for lift coefficients (C_L) between -0.2 and +0.2 and instability for C_L 0.2. It is estimated that a rearward shift of two percent of the wing coupled with a 20% increase in tail area will produce a stability margin (dC_m/dC_L) of approximately -1.5 percent. Referring to the wind tunnel model drawing, shown in Figure 3-34, it is seen that incorporating these changes would result in a configuration approaching that of the clipped delta which essentially verifies the earlier conclusions.

2) Clipped Delta Wing Location - Figures 3-35 and 3-36 show the effect of wing vertical location on the aerodynamic force and moment characteristics. For the high wing configuration there is no change in the lift curve slope, but the zero angle lift value changes from 0.25 to 0.22. There was a similar percentage reduction in the drag data, which resulted in no change in the maximum L/D. The moment data shows that zero angle moment is less negative and the slope of the moment curve indicates that the high wing shape is slightly less stable, at least at the low angles of attack.

3) Clipped Delta Wing Thickness - Figures 3-37 and 3-38 present the lift and moment increments caused by a 9% thick wing installation. For this configuration also, there is no change in the lift curve slope but the zero angle lift value decreased to 0.19. The zero angle drag change for this configuration was from .037 to .032, which with the reduced induced drag resulted in an increase in L/D from 7.4 to 7.8. The zero angle moment value is less negative, but the slope of the curve shows that this configuration is more stable.

4) Clipped Delta Root Fairing - Figures 3-39 and 3-40 show effects of the addition of a leading edge root fairing. The lift curve slope and the zero angle lift were unchanged. However, there was a 3% increase in vehicle drag which caused a corresponding decrease in maximum L/D; i.e. from 7.4 to 7.2.

WING-TAIL CARRIER CONFIGURATION

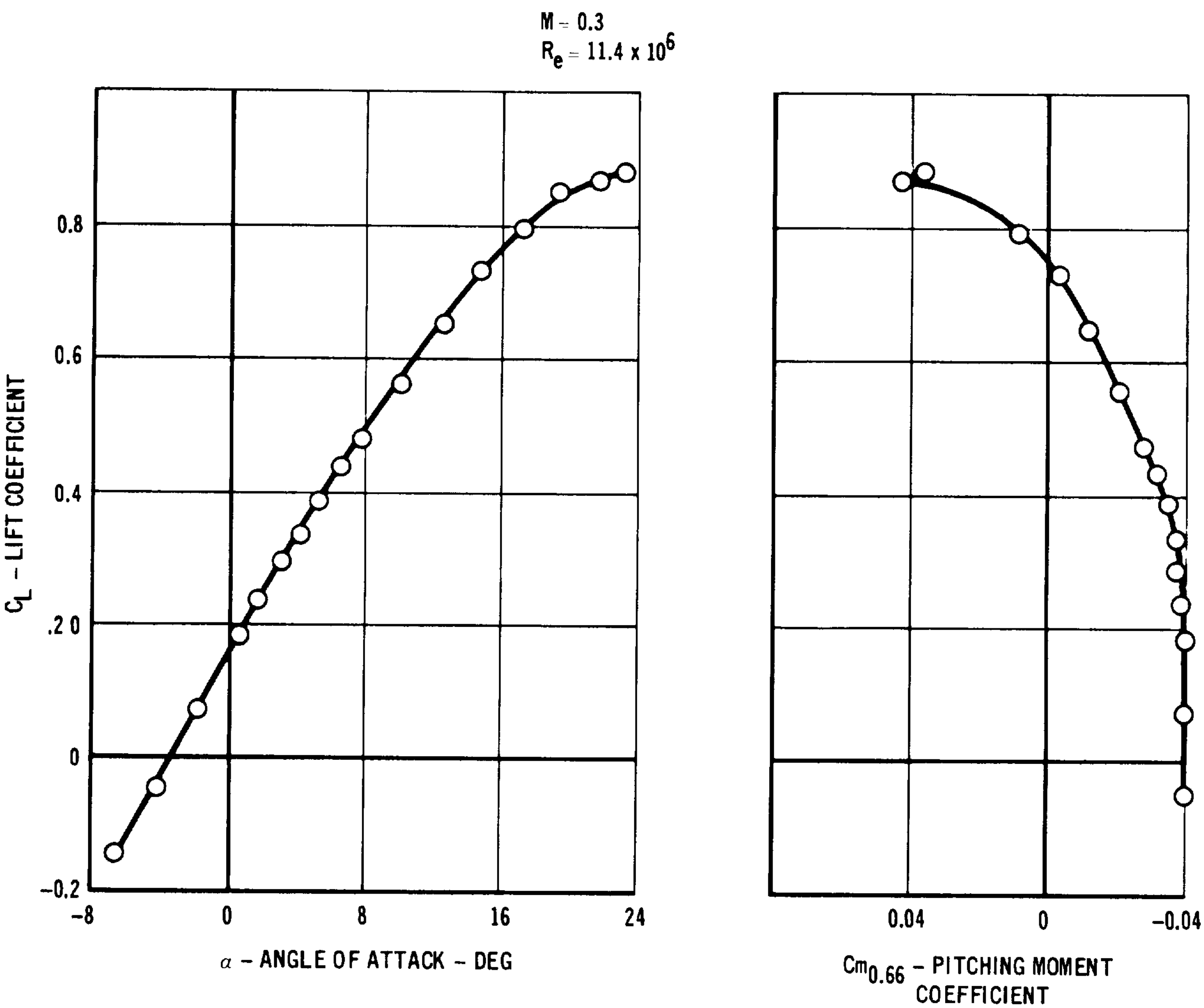


Figure 3-31

WING-TAIL CARRIER CONFIGURATION

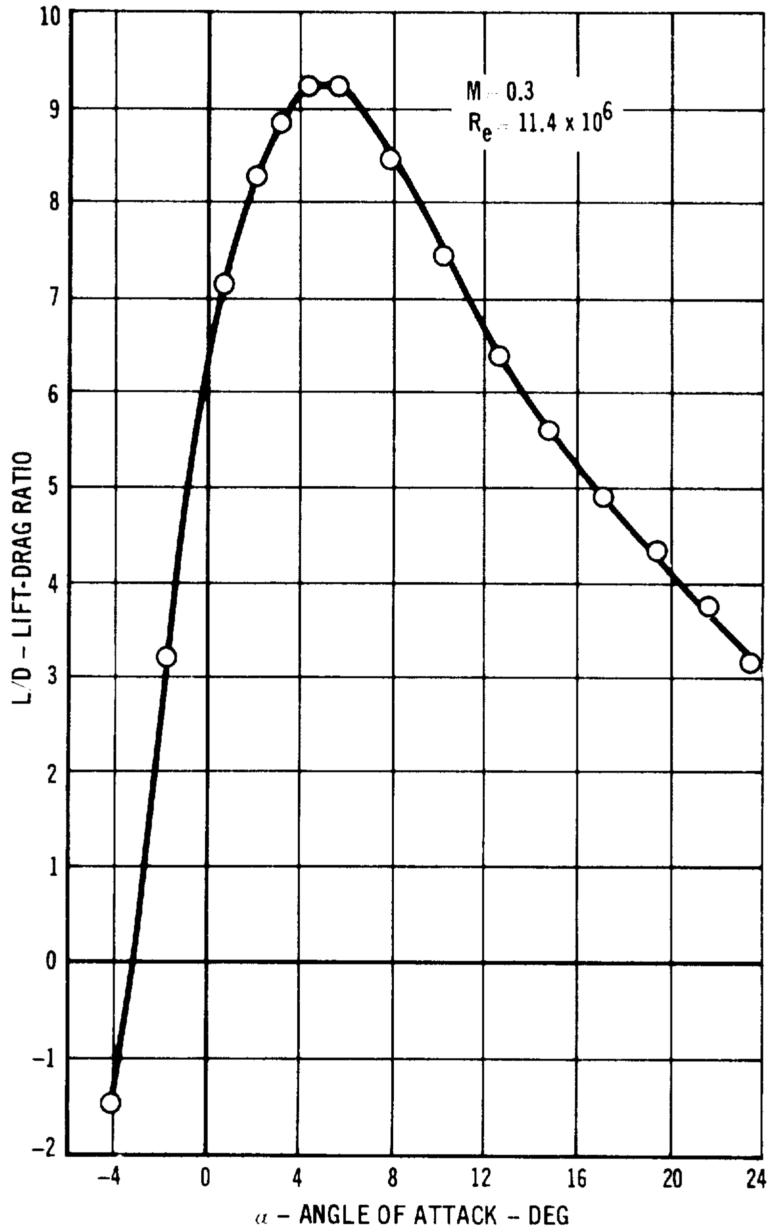


Figure 3-32

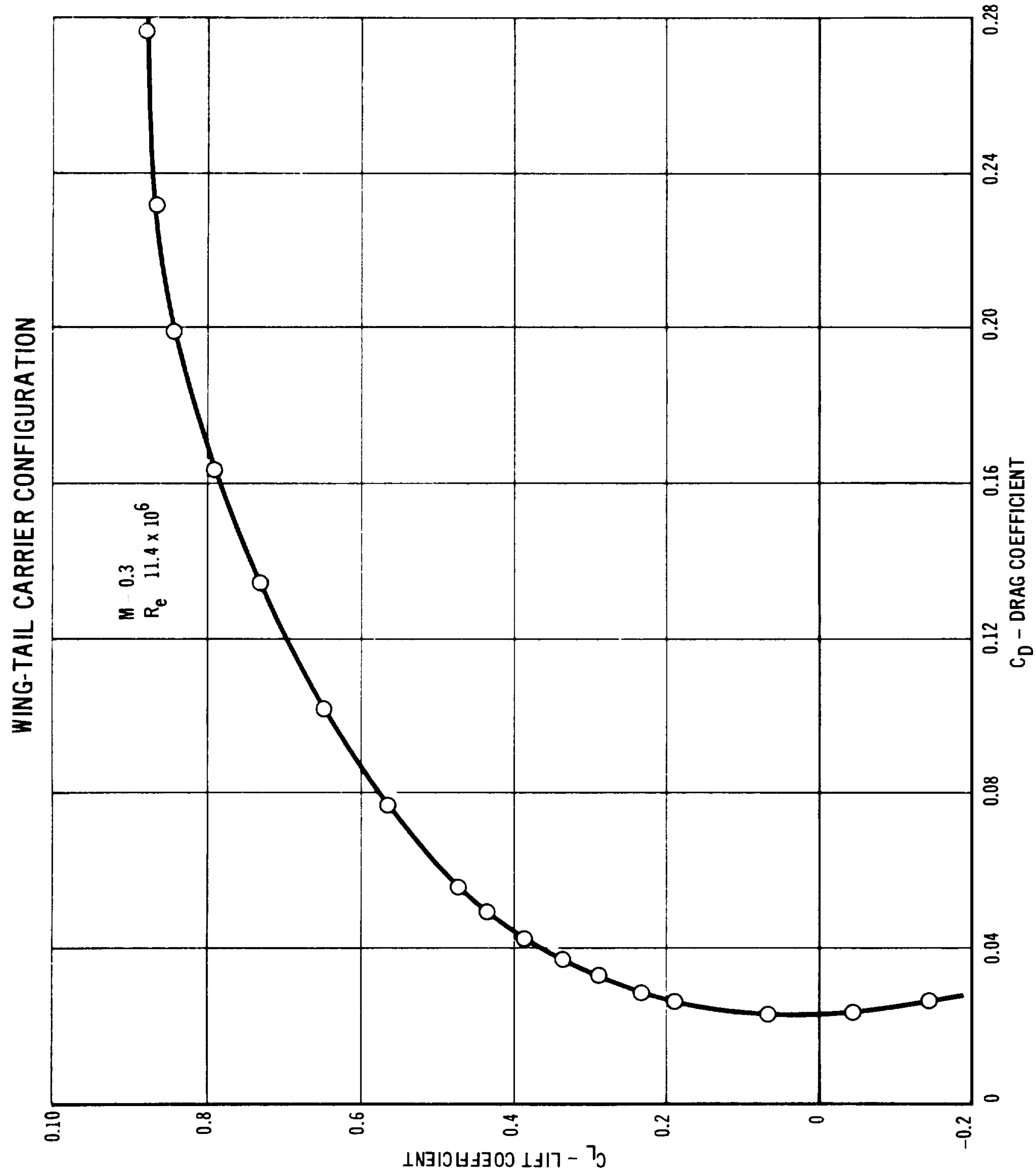

WING-TAIL CARRIER CONFIGURATION
M = 0.3
R_e = 11.4 x 10^6
C_D – DRAG COEFFICIENT
C_L – LIFT COEFFICIENT
0
0.04
0.08
0.12
0.16
0.20
0.24
0.28
1.0
0.8
0.6
0.4
0.2
0
-0.2

Figure 3-33

WING-TAIL CARRIER CONFIGURATION

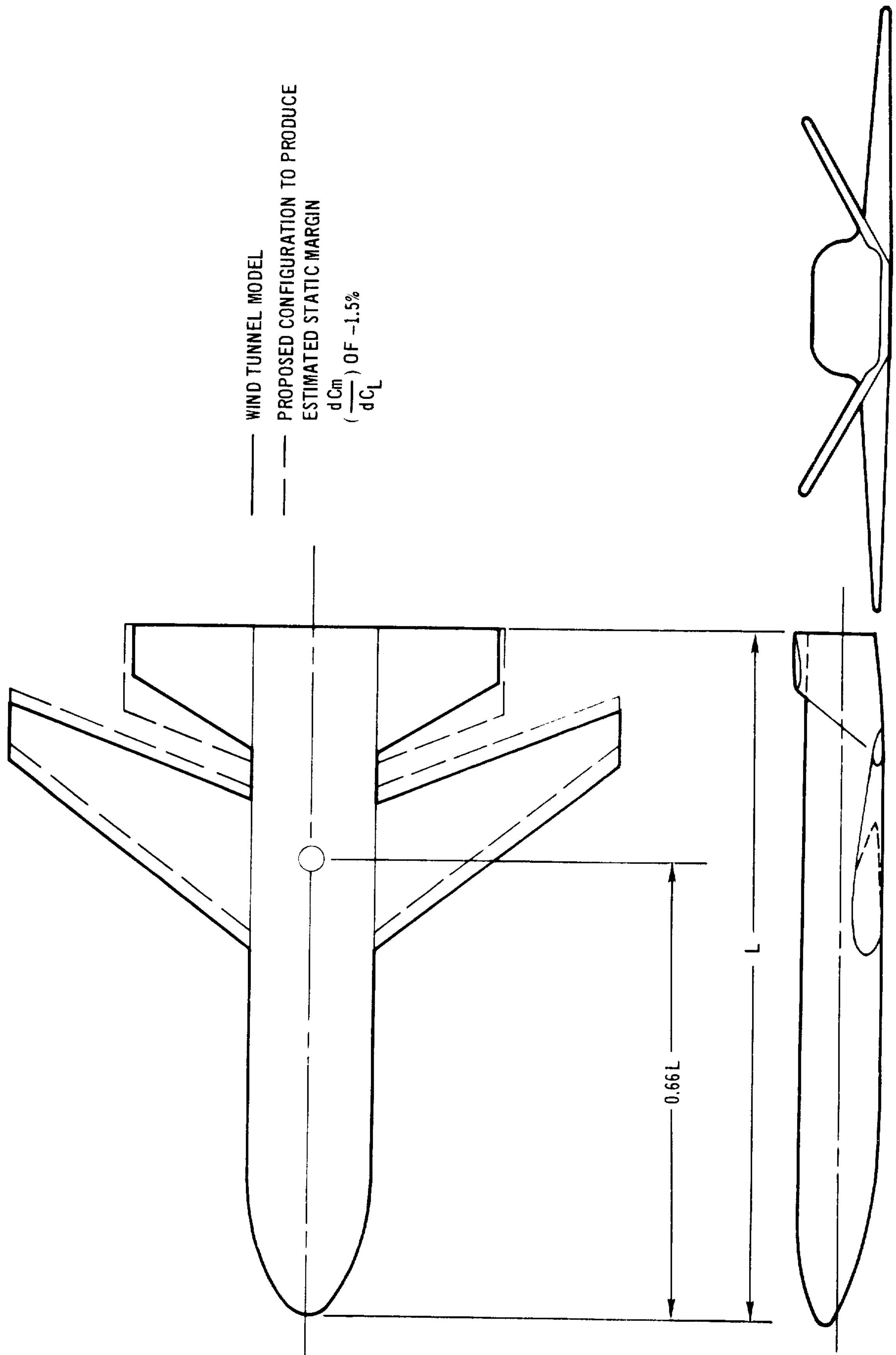


Figure 3-34

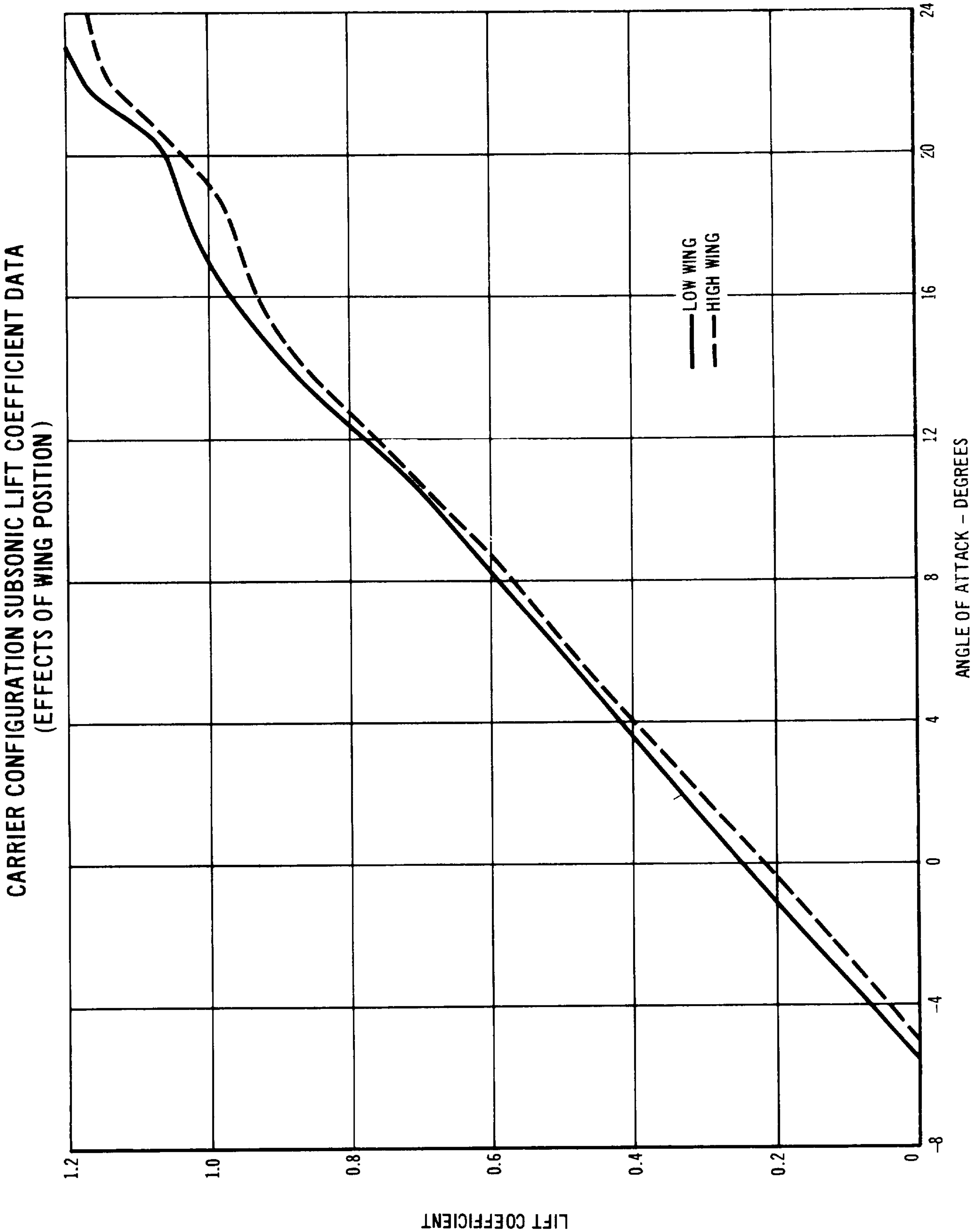

CARRIER CONFIGURATION SUBSONIC LIFT COEFFICIENT DATA
(EFFECTS OF WING POSITION)
LOW WING
HIGH WING
ANGLE OF ATTACK – DEGREES
LIFT COEFFICIENT
-8
-4
0
4
8
12
16
20
24
0
0.2
0.4
0.6
0.8
1.0
1.2

Figure 3-35

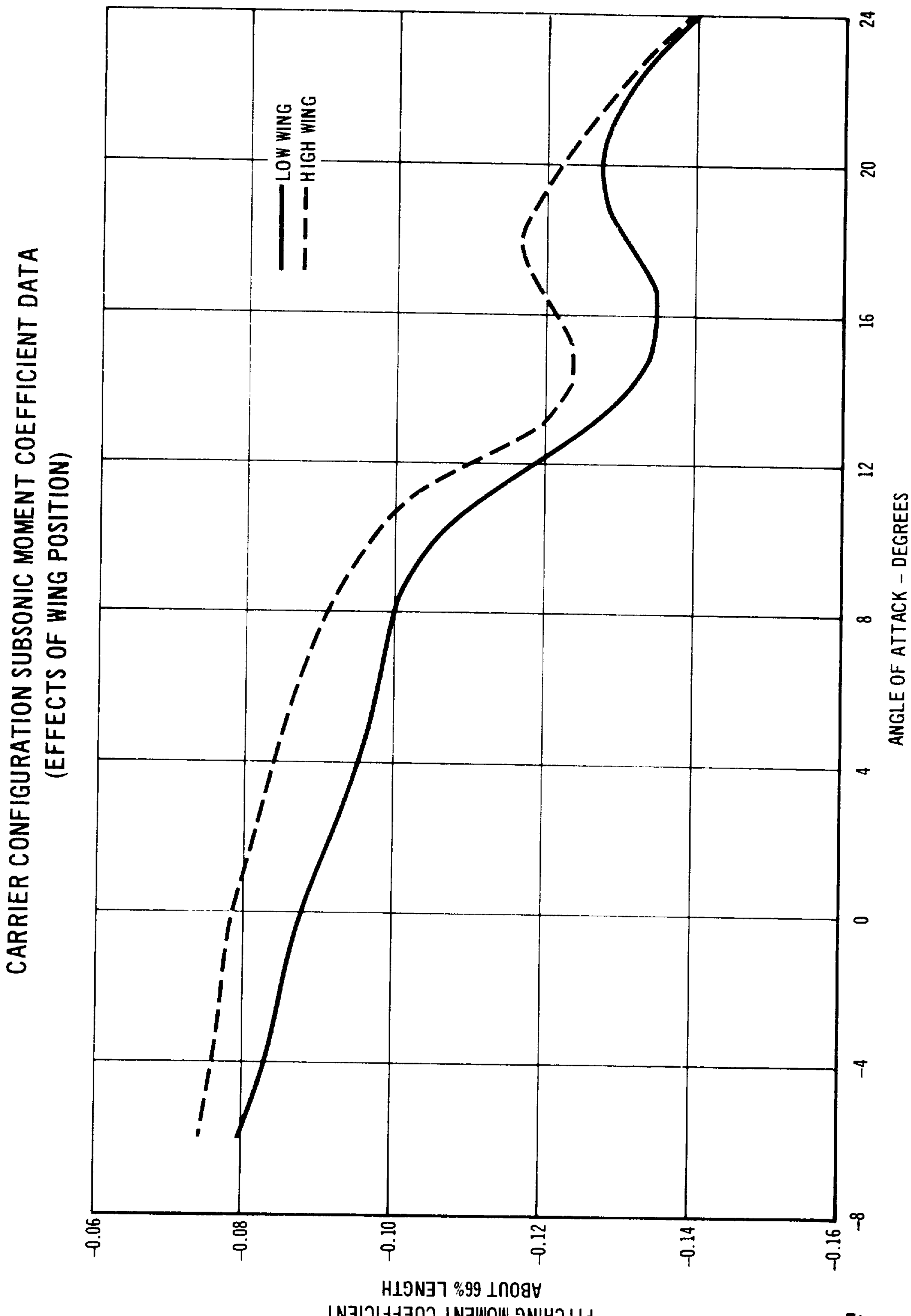

CARRIER CONFIGURATION SUBSONIC MOMENT COEFFICIENT DATA
(EFFECTS OF WING POSITION)
LOW WING
HIGH WING
ANGLE OF ATTACK – DEGREES
-8
-4
0
4
8
12
16
20
24
PITCHING MOMENT COEFFICIENT
ABOUT 66% LENGTH
-0.06
-0.08
-0.10
-0.12
-0.14
-0.16

Figure 3-36

CARRIER CONFIGURATION SUBSONIC LIFT COEFFICIENT DATA
(EFFECTS OF WING THICKNESS)

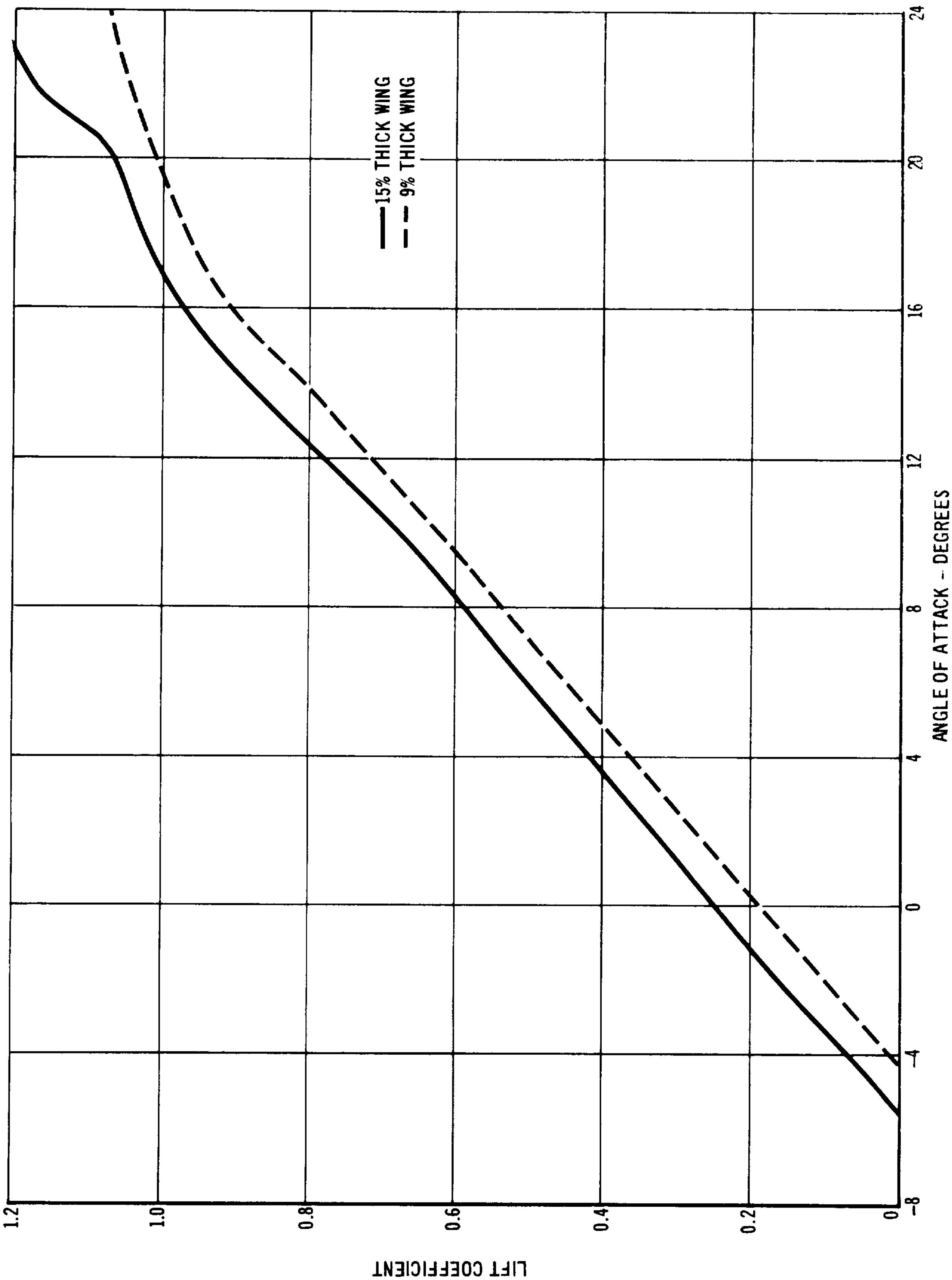


Figure 3-37

CARRIER CONFIGURATION SUBSONIC MOMENT COEFFICIENT DATA
(EFFECTS OF WING THICKNESS)

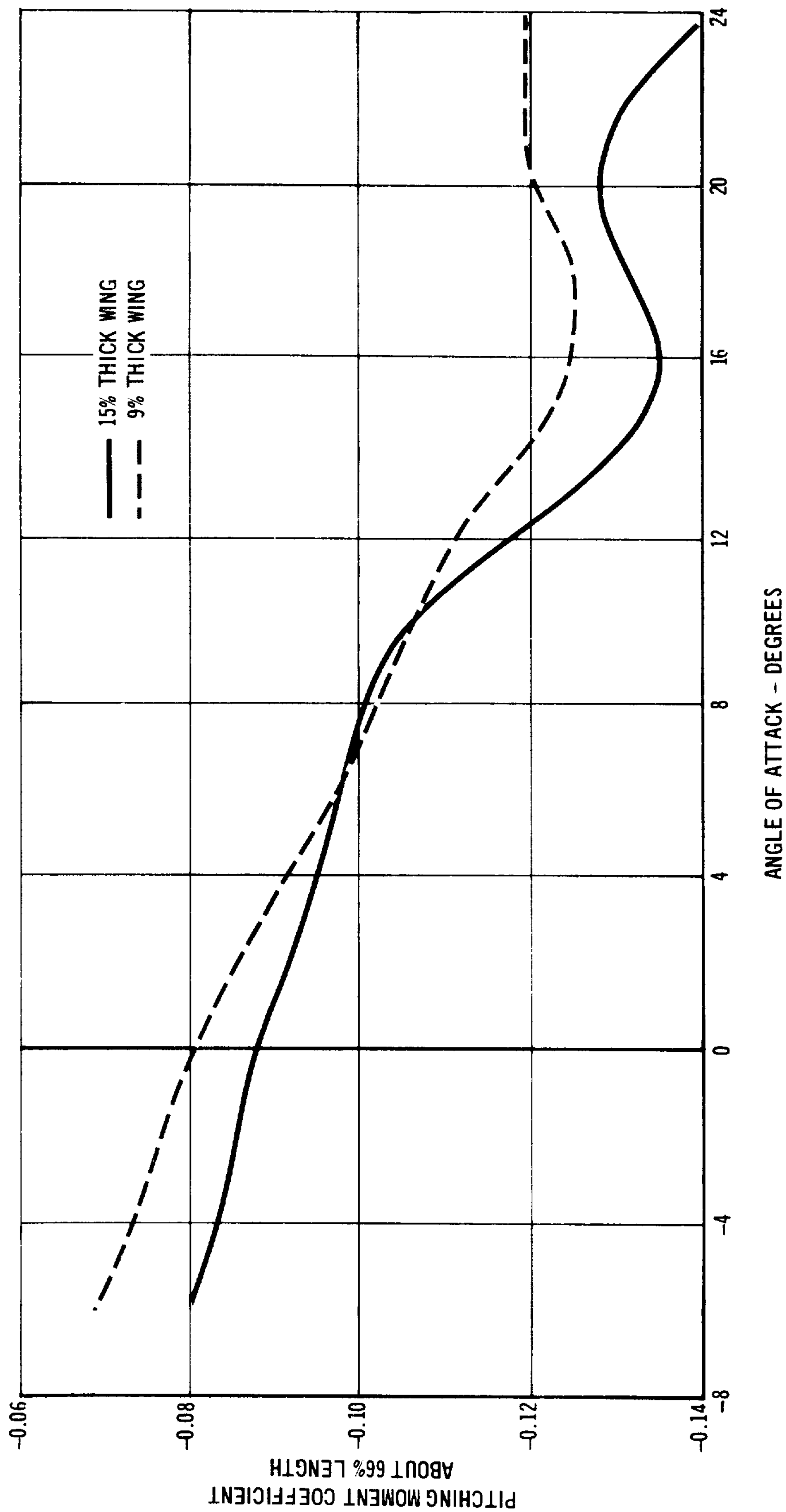


Figure 3-38

CARRIER CONFIGURATION SUBSONIC LIFT COEFFICIENT DATA
(EFFECTS OF WING FAIRING)

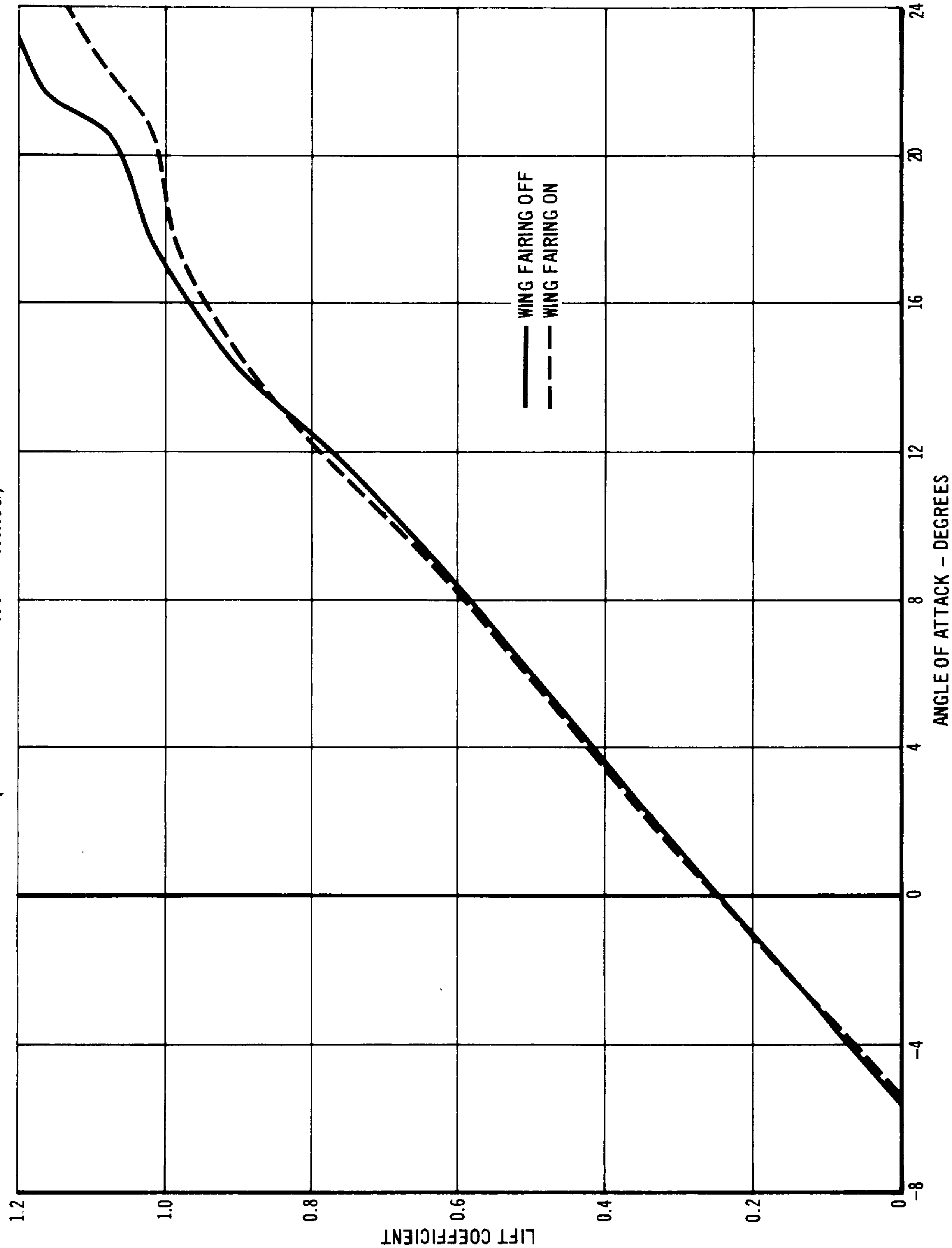


Figure 3-39

CARRIER CONFIGURATION SUBSONIC MOMENT COEFFICIENT DATA
(EFFECTS OF WING FAIRING)

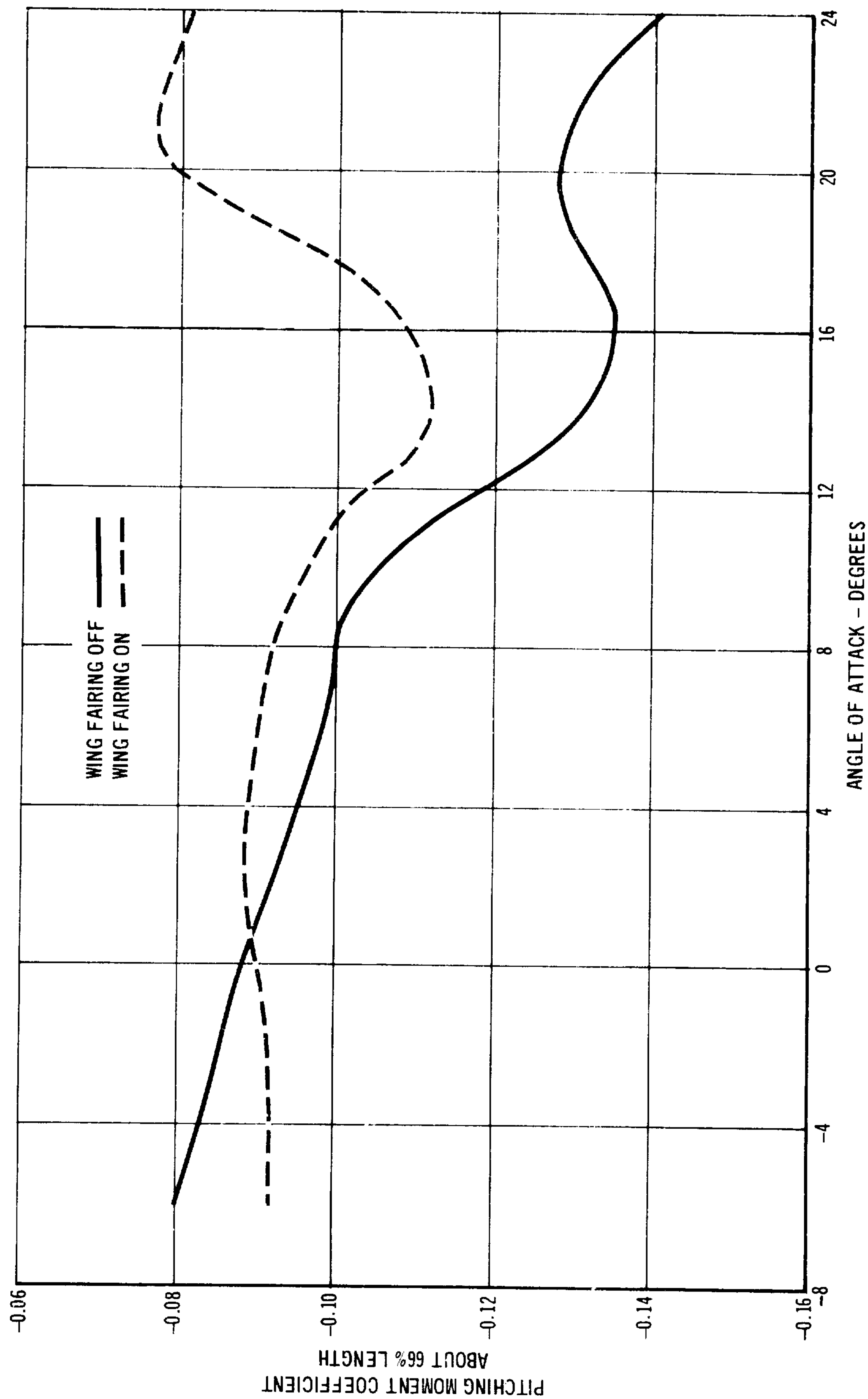


Figure 3-40

The zero moment value is essentially the same as the fairing off configuration but the slope change indicates that this configuration is less stable.

In summary, the clipped delta showed much better stability characteristics than the wing-tail configuration and the limited test data for the clipped delta indicate relative insensitivity of stability to the configuration variables which were tested. A maximum change of 5 percent improvement in L/D max was noted for the thin wing (9 percent) compared with the baseline configuration (15 percent).

- Hypersonic Aerodynamics - Figures 3-41 through 3-44 shows the hypersonic data for the Carrier configuration tested in Langley Research Center Continuous Flow Hypersonic Tunnel. Two configurations were tested at Mach number 10.4; i.e., the vehicle with the elevons at zero deflection angle, and the vehicle with the elevon control surfaces removed. A hypersonic estimate for the vehicle with zero control deflection has been superimposed on the test data for comparison purposes. The estimate was generated at the testing facility with the Hypersonic Arbitrary Body program, utilizing standard Newtonian theory (maximum pressure coefficient of 2.0) on the windward surfaces and Prandtl-Meyer expansion techniques on the leeward surfaces. Examination of the lift data indicates that the zero angle of attack lift value is -0.011 and slope of the lift curve is 0.010 per degree. The hypersonic estimate yields a value of -0.010 for the zero angle value and predicts a lift curve slope of approximately 0.008 per degree. The drag curve shows that the zero angle-of-attack drag data is 0.066, while the estimate predicts a value of 0.069. The figure which presents the zero elevon Lift-Drag ratio indicates that the maximum value of L/D is 1.6 at 18° angle of attack. The estimated L/D shows favorable agreement at all but the lowest angles of attack and predicts a maximum value of 1.62. The moment curve shows that the Carrier will trim at 20° angle of attack, for the chosen moment reference point. The data indicates that the zero angle-of-attack pitching moment coefficient is 0.0071, while the estimated curve predicts a value of 0.0068. The force curves and the moment curve show that hypersonic estimates agree well at the low angles of attack but tend to under predict the values at the higher angles. The pitching moment is the most sensitive coefficient, and therefore shows the largest percentage variation between the test data and the estimates. While the data shows the

CARRIER CONFIGURATION HYPERSONIC TEST DATA
LIFT COEFFICIENT

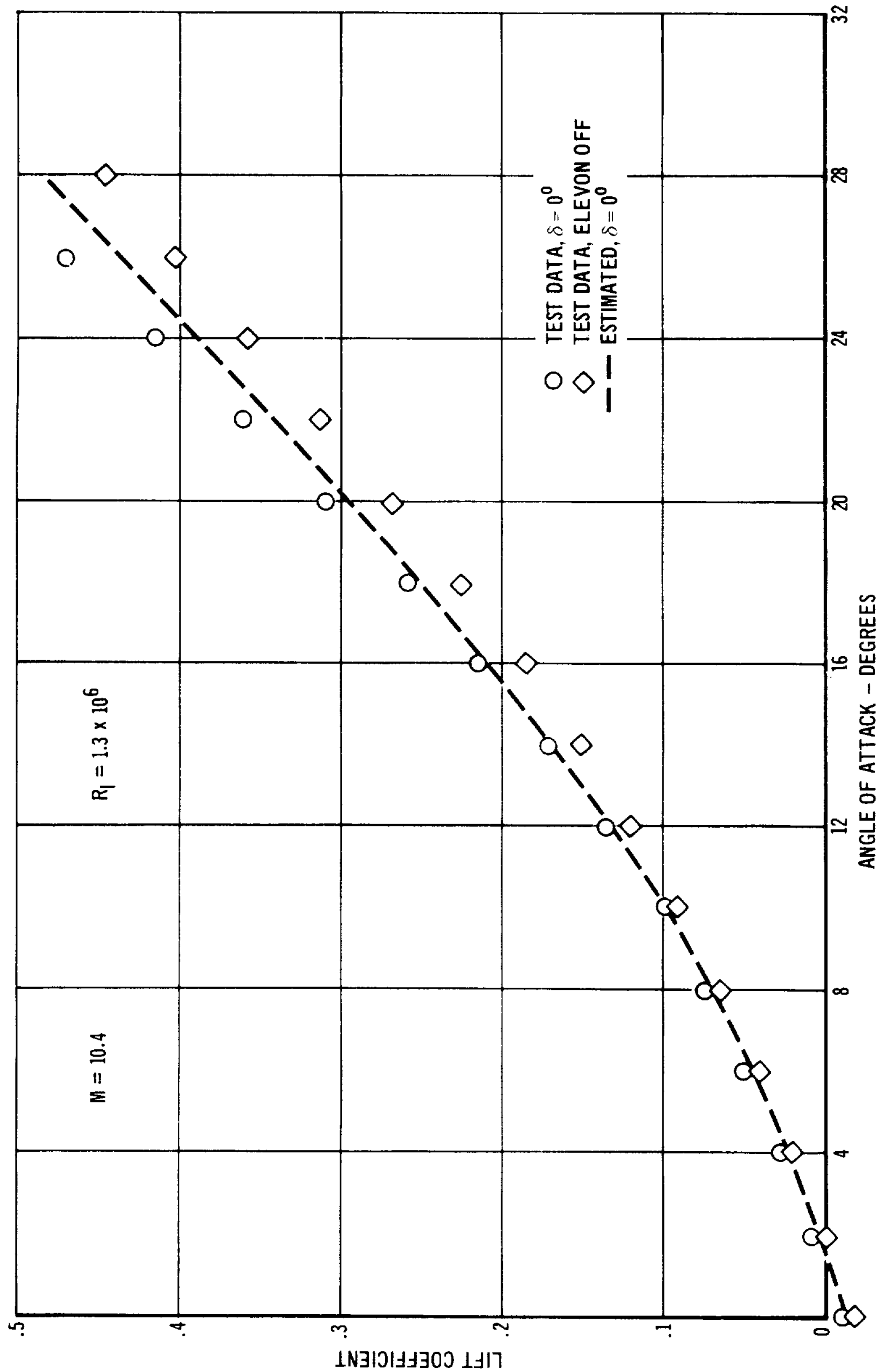


Figure 3-41

CARRIER CONFIGURATION HYPERSONIC TEST DATA
DRAG COEFFICIENT

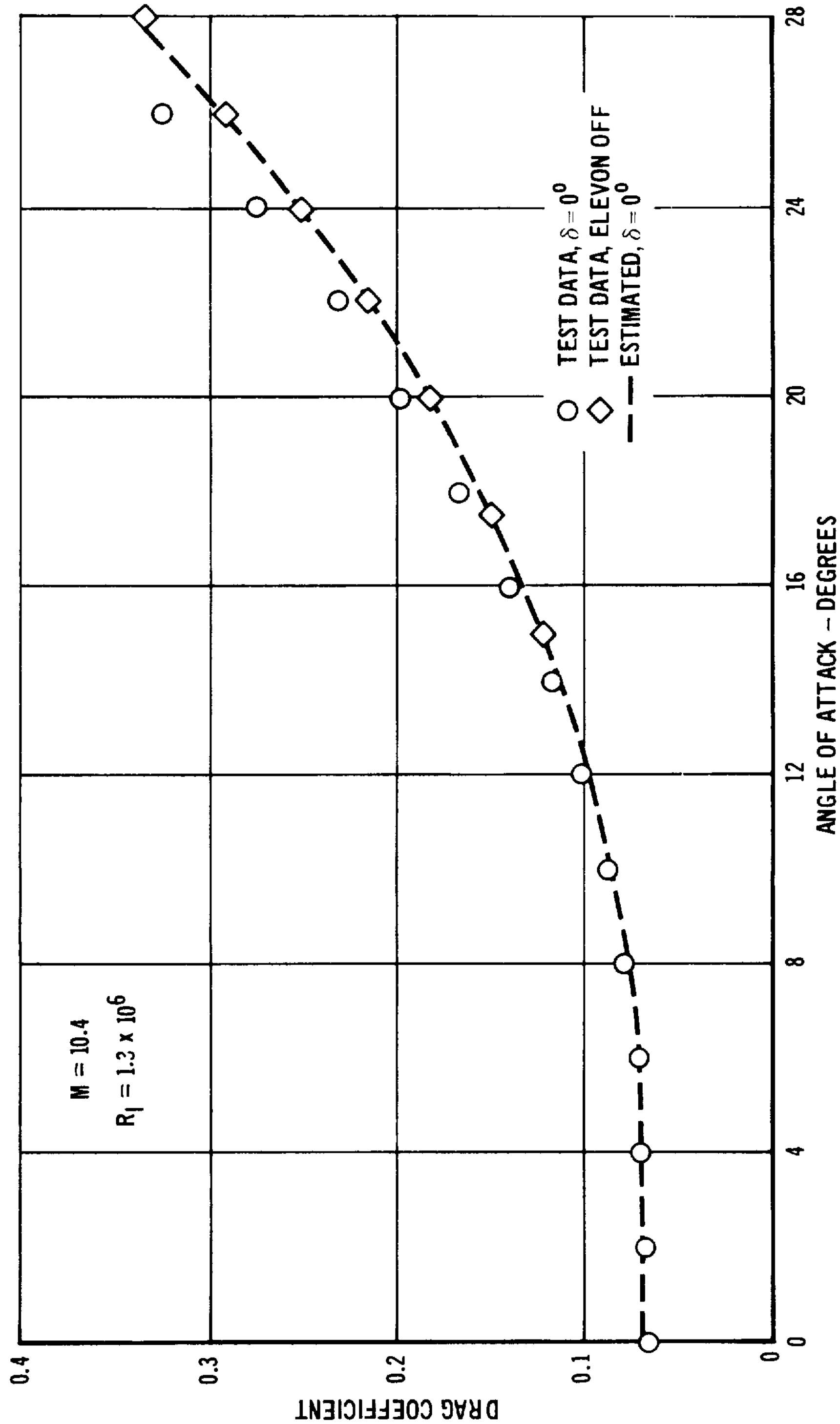
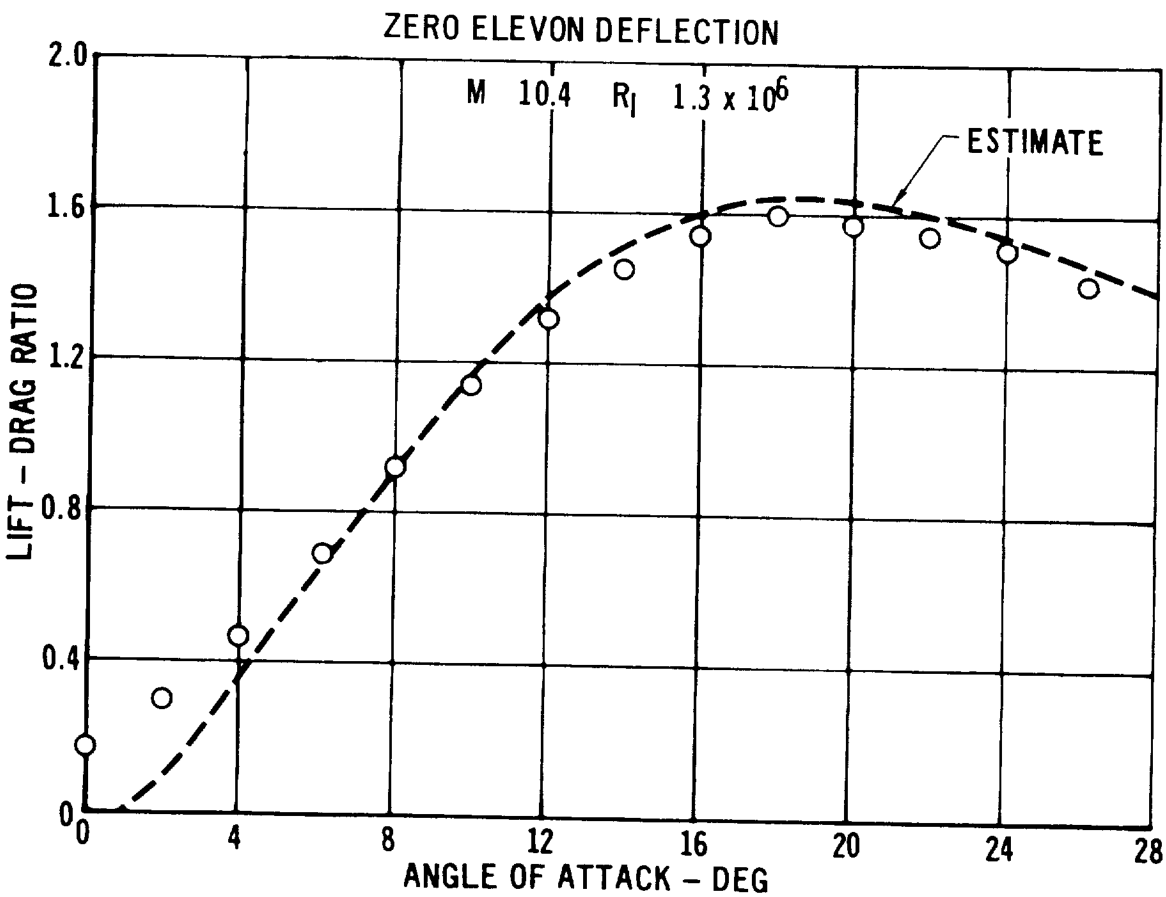


Figure 3–42

CARRIER CONFIGURATION
HYPERSONIC TEST DATA
LIFT DRAG RATIO



ILRVS-458 F

Figure 3-43

CARRIER CONFIGURATION HYPERSONIC TEST DATA
PITCHING MOMENT COEFFICIENT

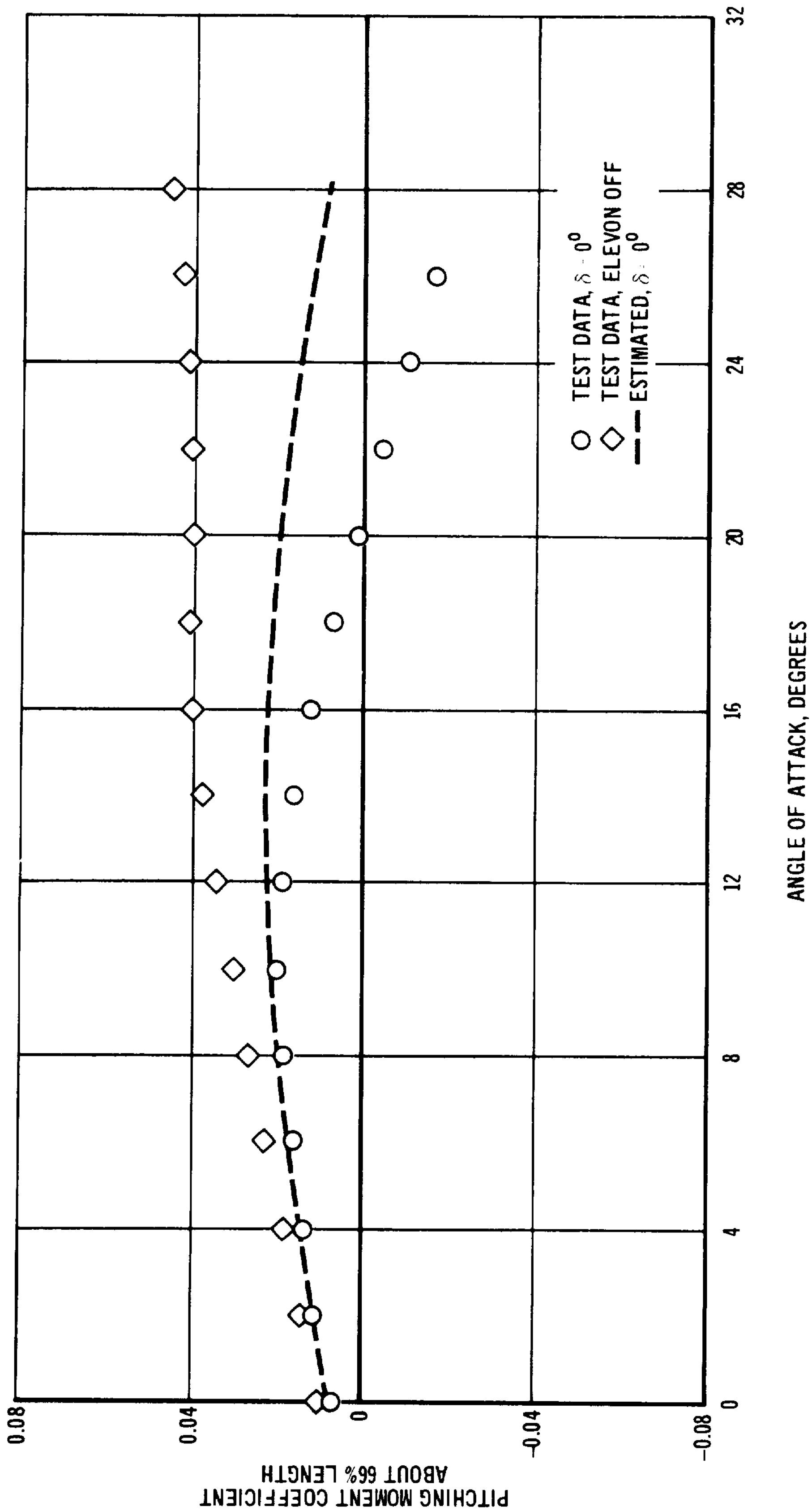


Figure 3-44

trim point at 20° angle of attack, the estimate indicates that trim will occur at 32°. The difference in trim angle of attack represents a 2% difference in center of pressure location.

One explanation for this difference is the fact that the Carrier vehicle is not immersed in a pure Newtonian flowfield. Thermographic test results at the Mach 10.4, 15° angle of attack condition, show that regions of the Carrier wing are subjected to increased heating rates. These regions are the results of shock wave interaction. Figure 3-49 shows a sketch of an estimated shock pattern which was used to explain the displacement of the thermographic test material. The presence of such a shock pattern would preclude the existence of a Newtonian flowfield and account for the fact that the data was higher than the estimates in this angle-of-attack range.

The wind tunnel model used to obtain low angle of attack aerodynamic characteristics of the Clipped Delta, was a 16 inch model, without wing filets and with a - 30 incidence lower wing. A 10 inch version of the same model was used to obtain higher angle of attack data. These models were tested at Mach 10.4 in the Langley Research Center Continuous Flow Hypersonic Tunnel.

The experimental hypersonic longitudinal stability and control characteristics of the Clipped Delta Carrier are presented in Figures 3-45 through 3-48. Since the high angle of attack data were acquired toward the end of the study time did not allow for complete analysis, nor change in the reference area and length. The reference area and vehicle length applicable to the previous data (Figures 3-36 through 3-39) are total vehicle projected planform area and overall vehicle length. The aerodynamic characteristics are "primed" to indicate this fact. To base lift and drag coefficients on wing theoretical area, a factor of 1.258 should be employed. Similarly if the pitching moment coeffficient is to be based on wing theoretical area and mean aerodynamic chord, the factor to be employed is 2.86.

A review of these data indicate that a maximum trimmed lift-to-drag ratio of 1.60 (16" model) is attainable without any control deflection. Furthermore, a control deflection of - 10° does not noticeably affect L/D levels. The vehicle is stable and trimmable at angles of attack above 20°. The discontinuity in

CLIPPED DELTA LONGITUDINAL CHARACTERISTICS
(PART 4)

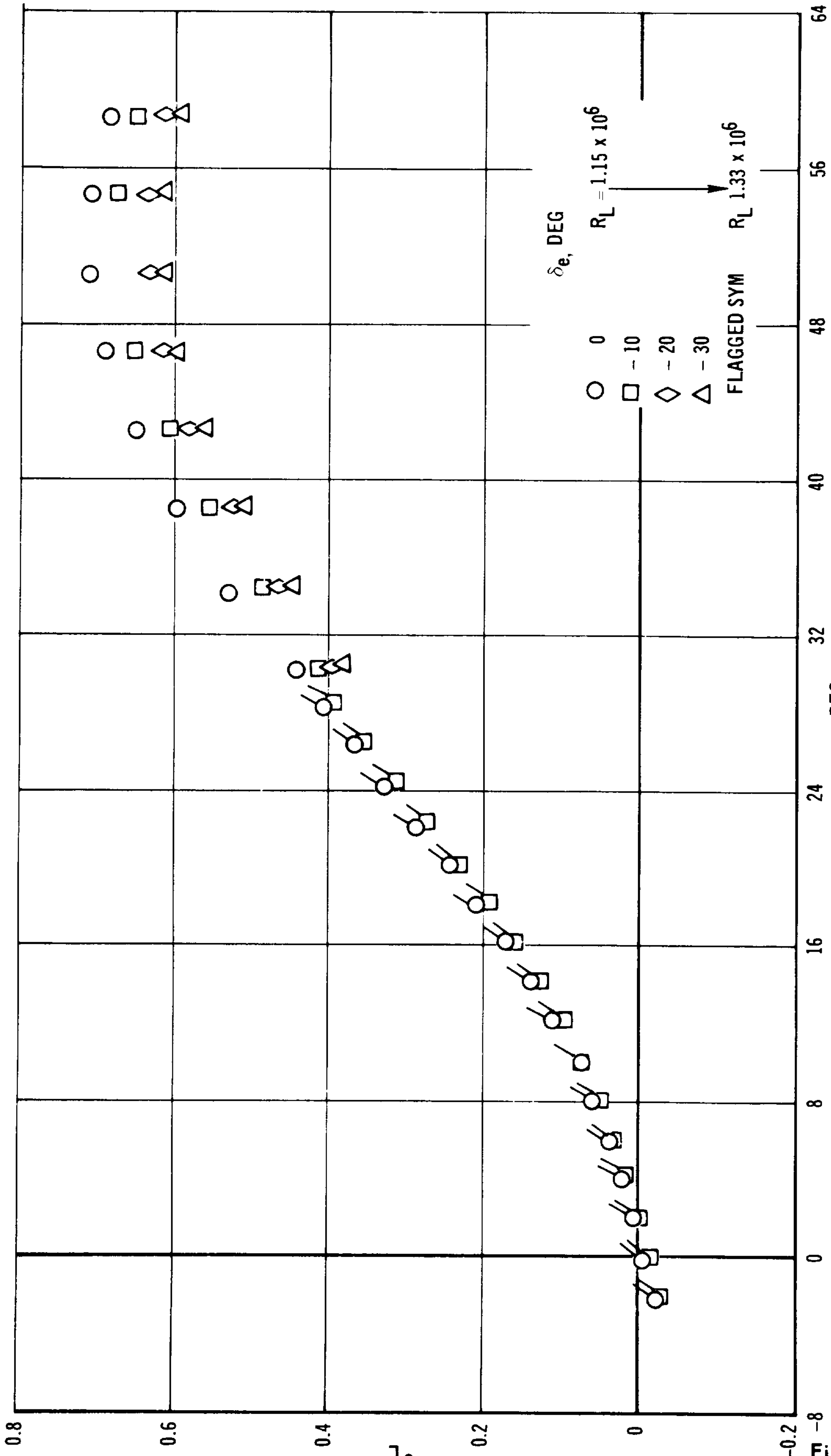


Figure 3-45

CLIPPED DELTA LONGITUDINAL CHARACTERISTICS
(PART 3)

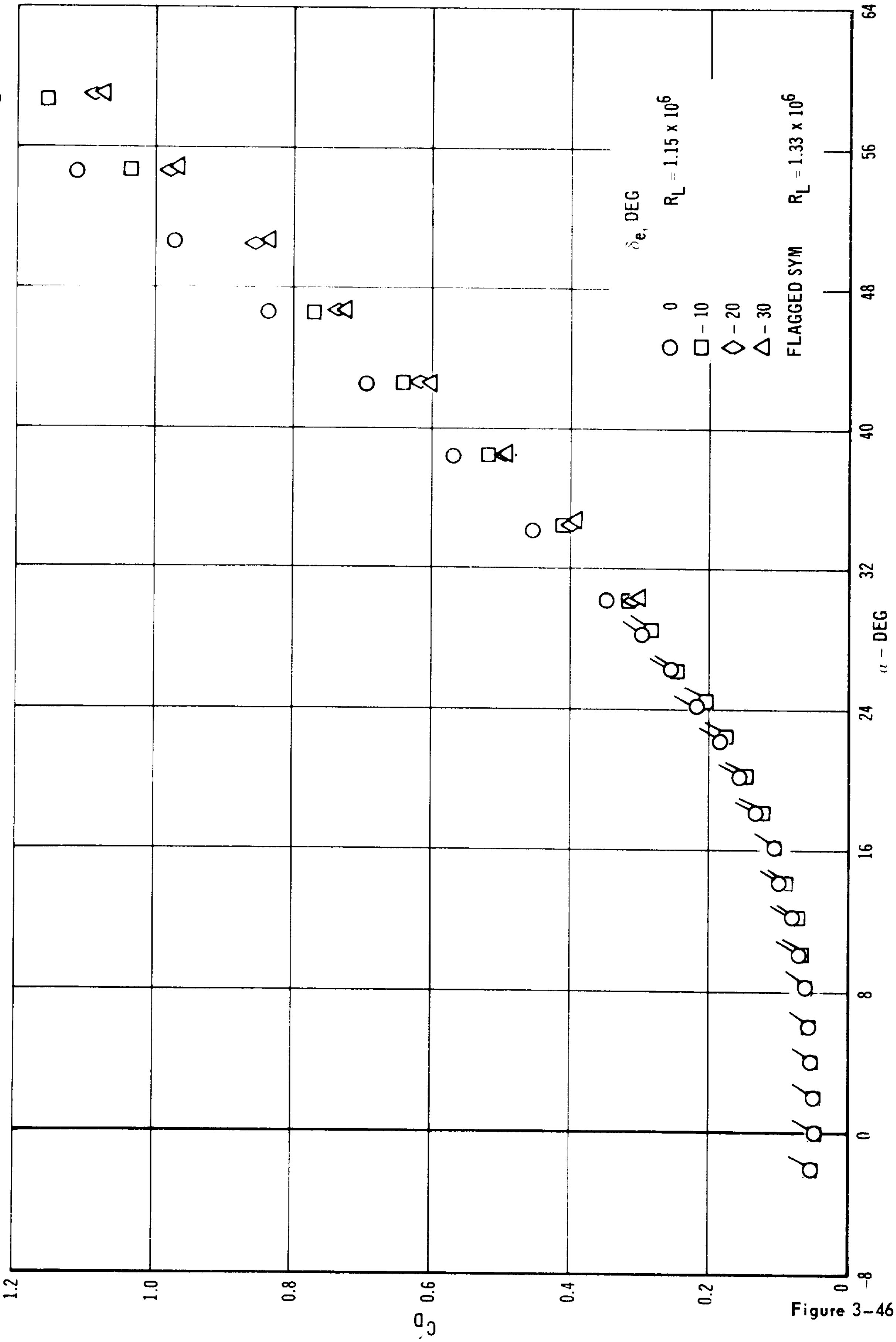


Figure 3-46

CLIPPED DELTA LONGITUDINAL CHARACTERISTICS
(PART 2)

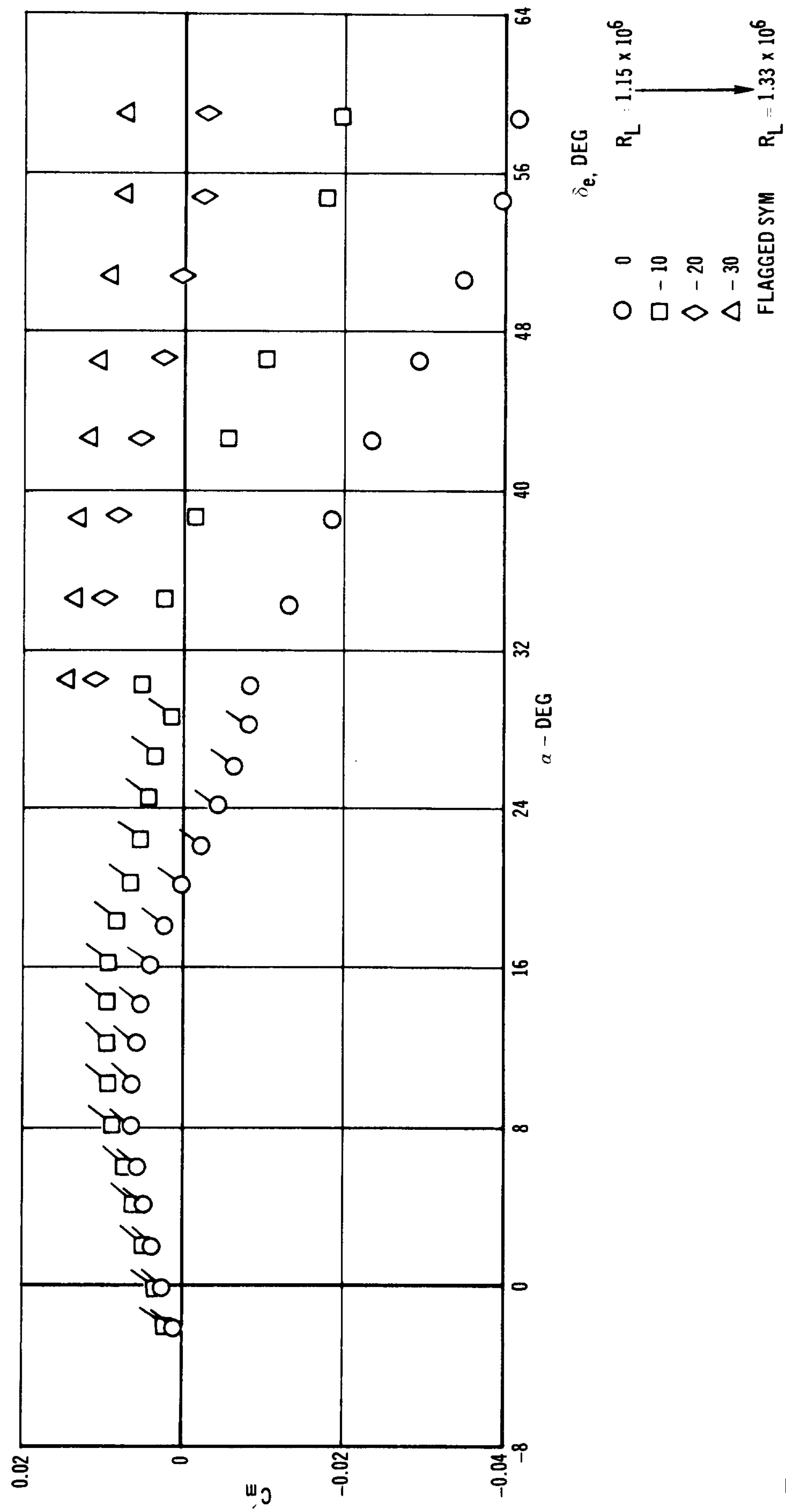


Figure 3–47

CLIPPED DELTA LONGITUDINAL CHARACTERISTICS
(PART 1)

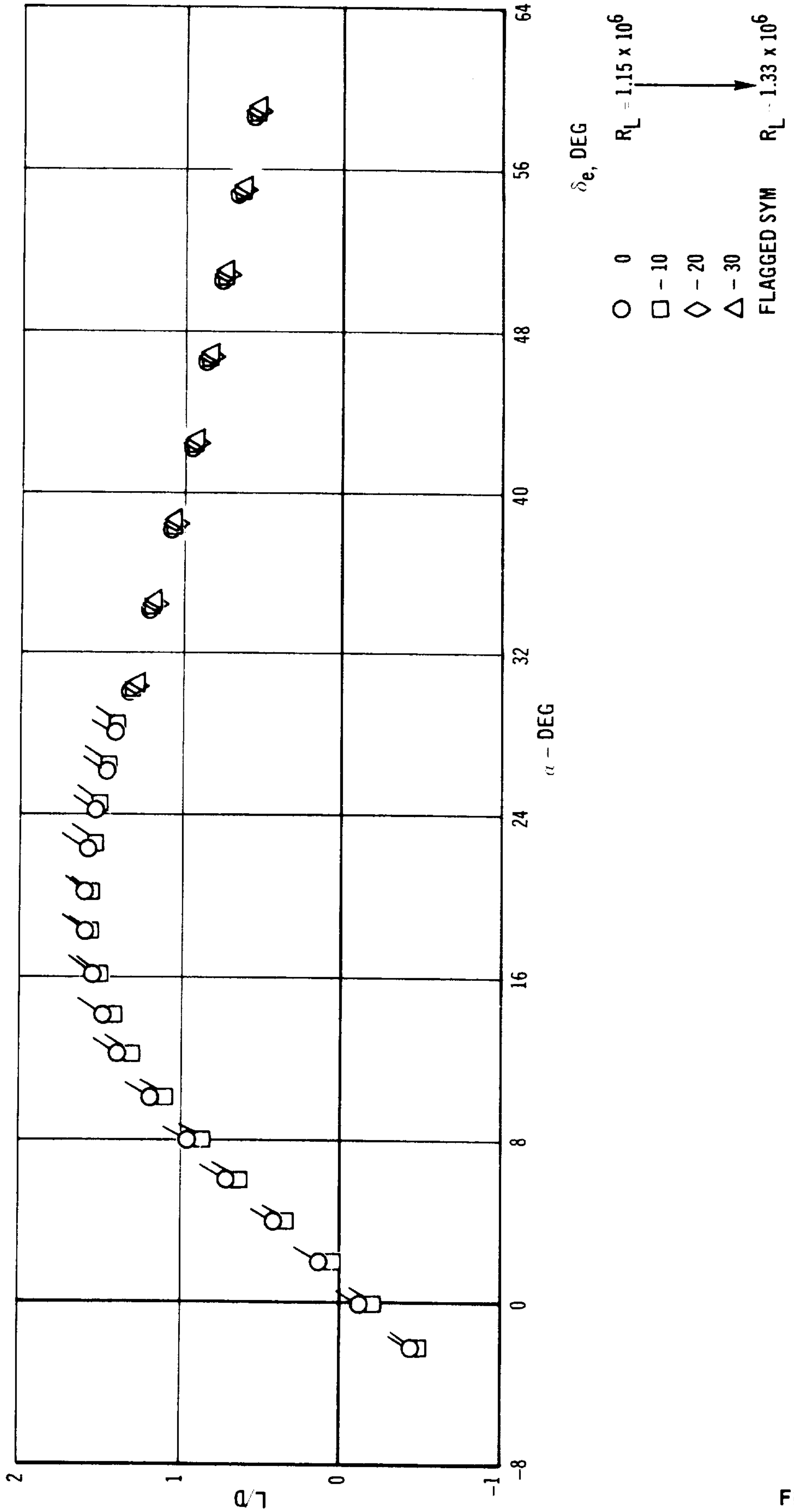


Figure 3-48

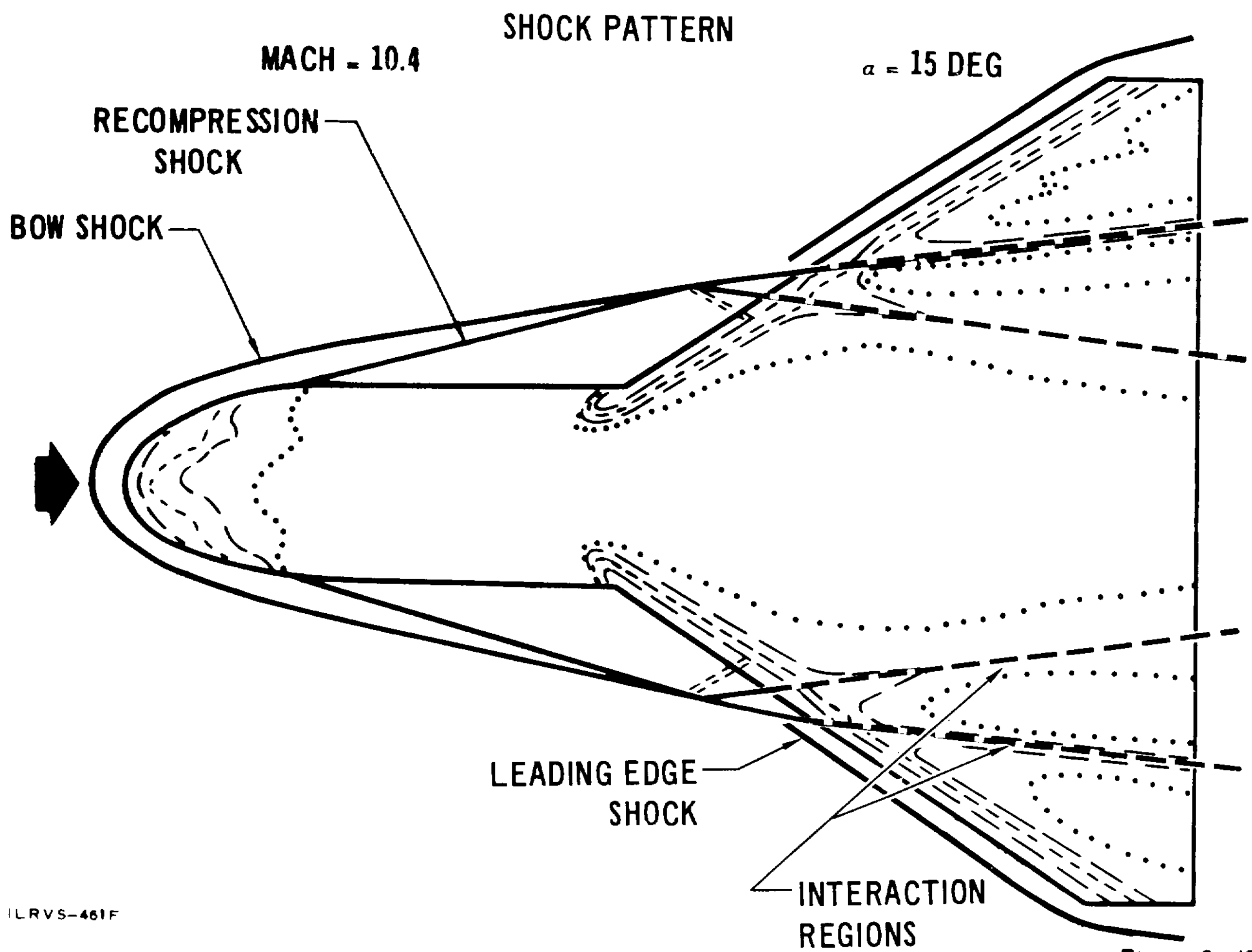


Figure 3-49

the data above 28° angle of attack is the result of sting installation from base mount to lee mount and is a result of flow interaction with the sting. This phenomenon has been observed before and is not a vehicle peculiar characteristic.

- <u>Tri-Sonic Aerodynamics</u> - In addition to the Carrier test data and the Hypersonic Arbitrary Body estimates, a series of untrimmed estimated values were included to provide an idea of the effect of Mach number variation on the aerodynamic characteristics of the vehicle. These estimates appear in Figures 3-50 through 3-60. The majority of these estimated characteristics were generated with hand calculations which were based on published test data for similar type aircraft configurations. On several of the figures, Carrier test data was incorporated at the appropriate Mach number. These calculations were based on the methods outlined in References 2 and 3 supported by the test data presented in References 4 through 7. In addition, LRC unpublished subsonic test data were used to substantiate the estimates for the Clipped Delta configuration.

CARRIER CONFIGURATION
VARIATION OF LIFT-CURVE SLOPE
WITH MACH NUMBER
ZERO ELEVON DEFLECTION

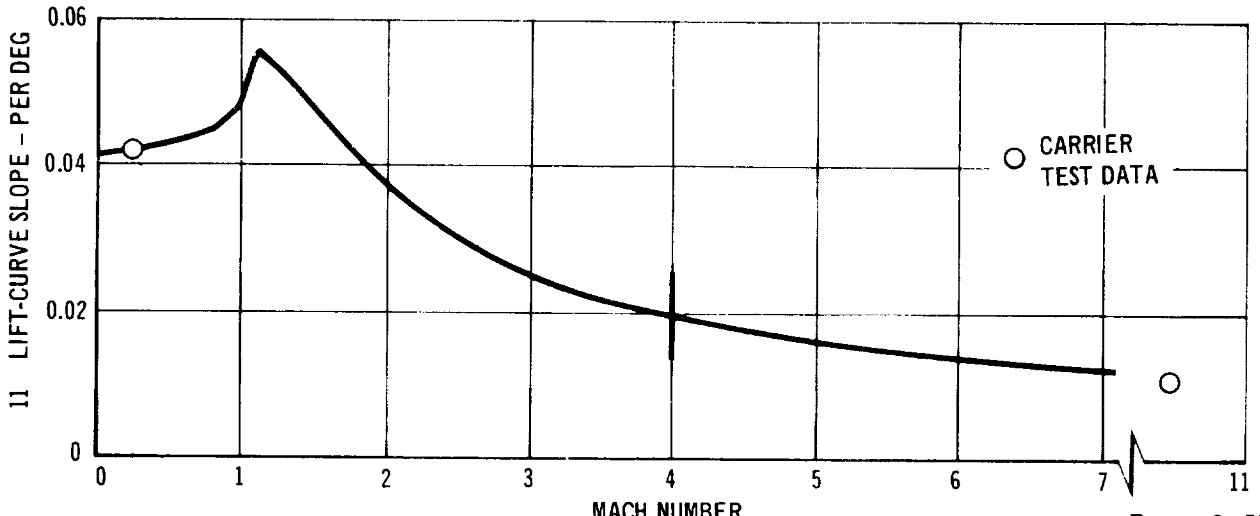


Figure 3–50

CARRIER CONFIGURATION
VARIATION OF MOMENT-CURVE SLOPE
WITH MACH NUMBER
ZERO ELEVON DEFLECTION

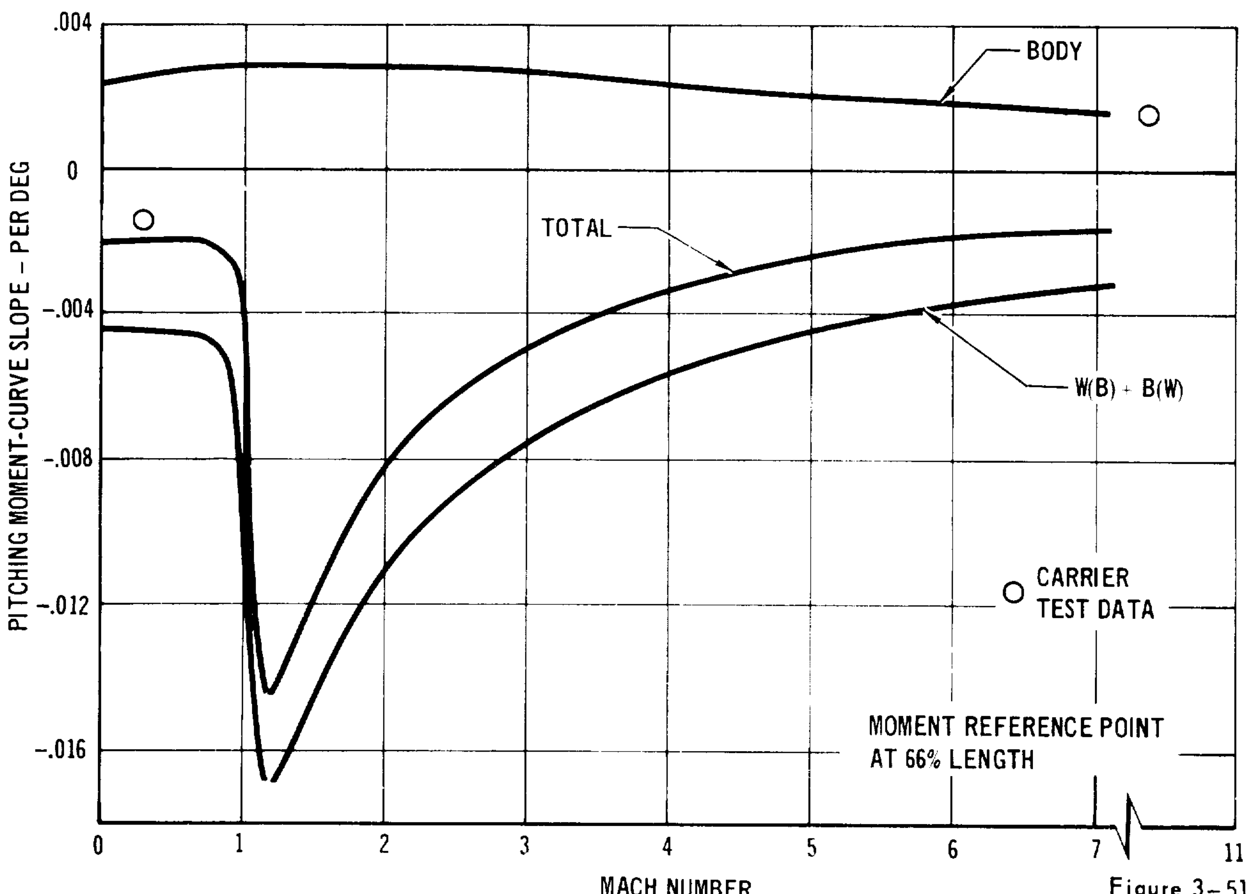


Figure 3–51

CARRIER CONFIGURATION
VARIATION OF LONGITUDINAL STATIC STABILITY MARGIN
WITH MACH NUMBER
ZERO ELEVON DEFLECTION

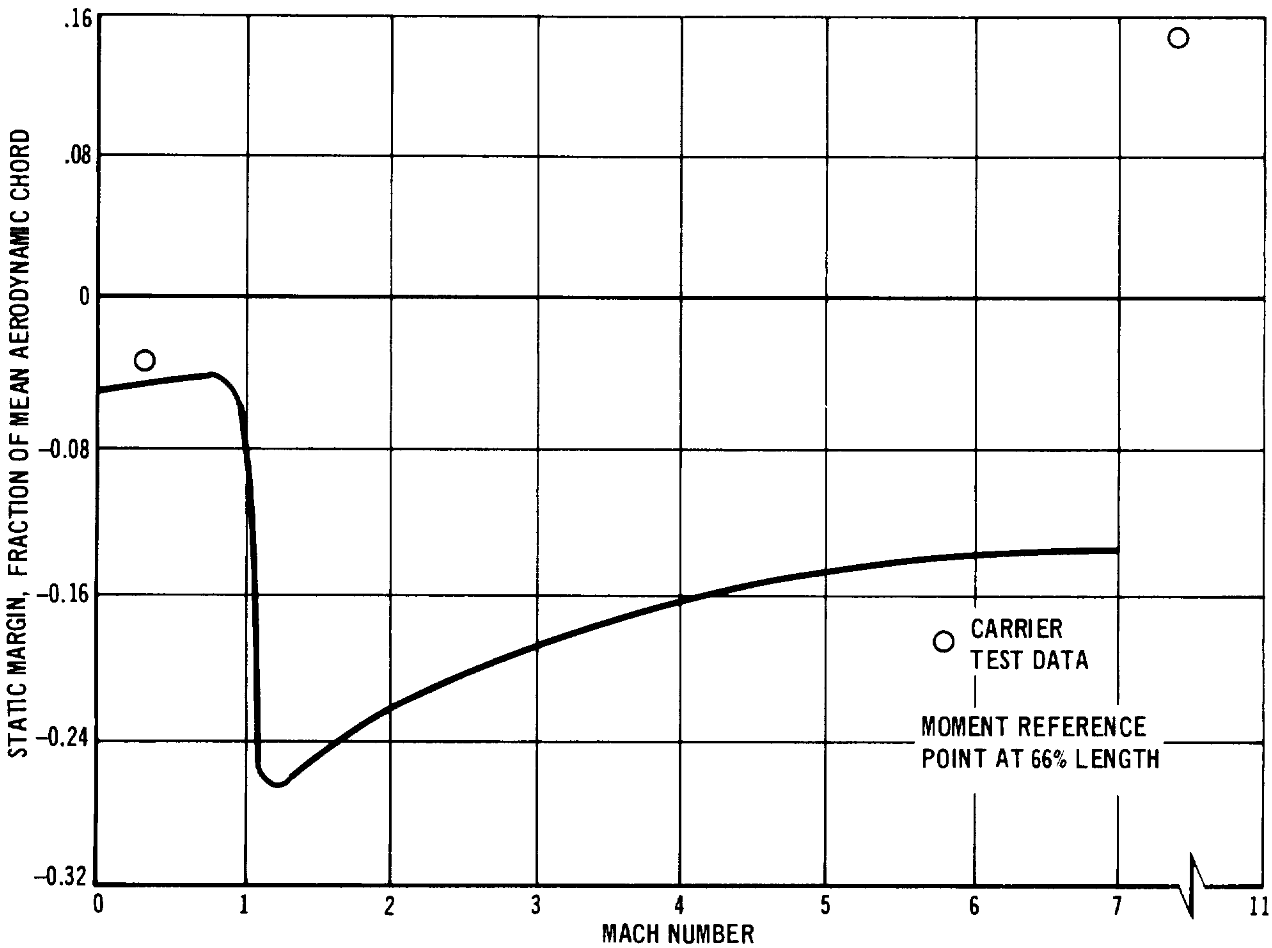


Figure 3-52

CARRIER CONFIGURATION
VARIATION OF MINIMUM DRAG COEFFICIENT
WITH MACH NUMBER
ZERO ELEVON DEFLECTION

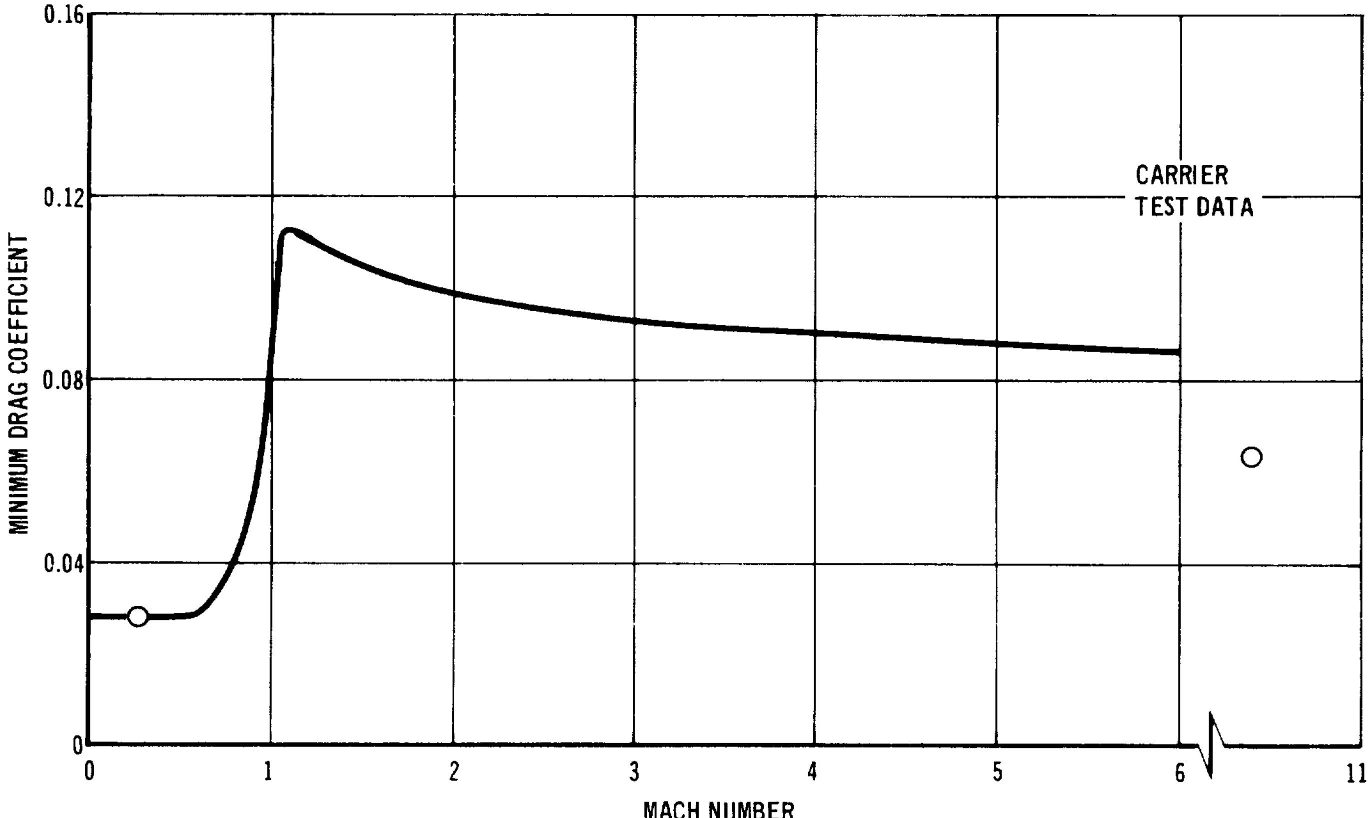


Figure 3-53

CARRIER CONFIGURATION
VARIATION OF DRAG-DUE-TO-LIFT PARAMETER
WITH MACH NUMBER
ZERO ELEVON DEFLECTION

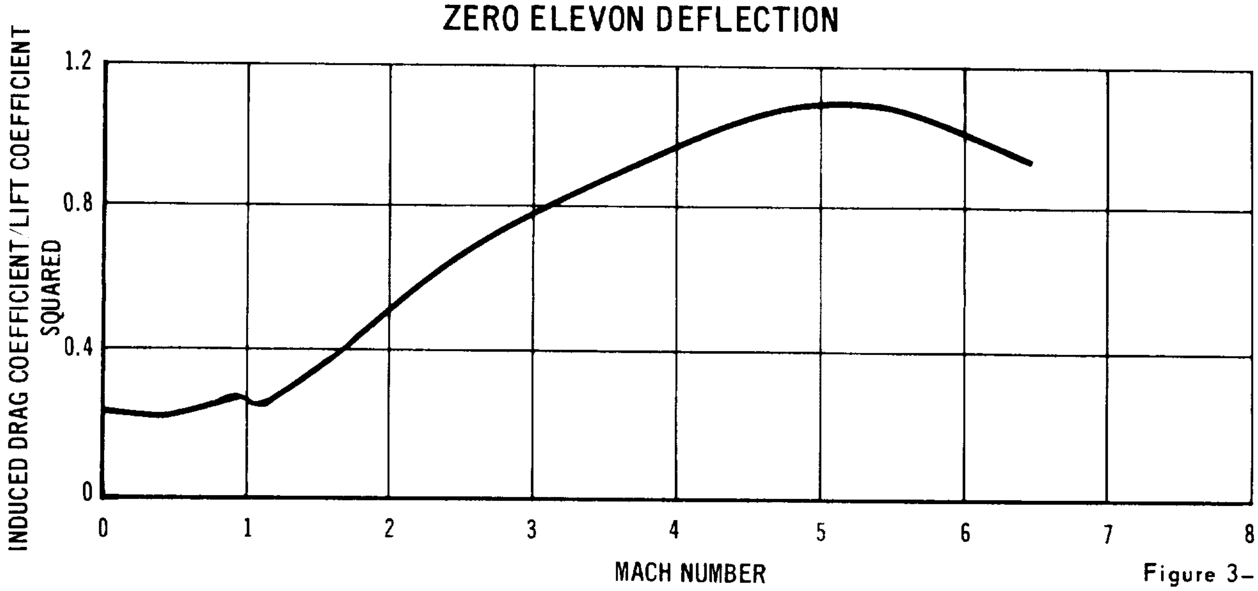


Figure 3-54

CARRIER CONFIGURATION
VARIATION OF MAXIMUM LIFT-TO-DRAG RATIO
WITH MACH NUMBER
ZERO ELEVON DEFLECTION

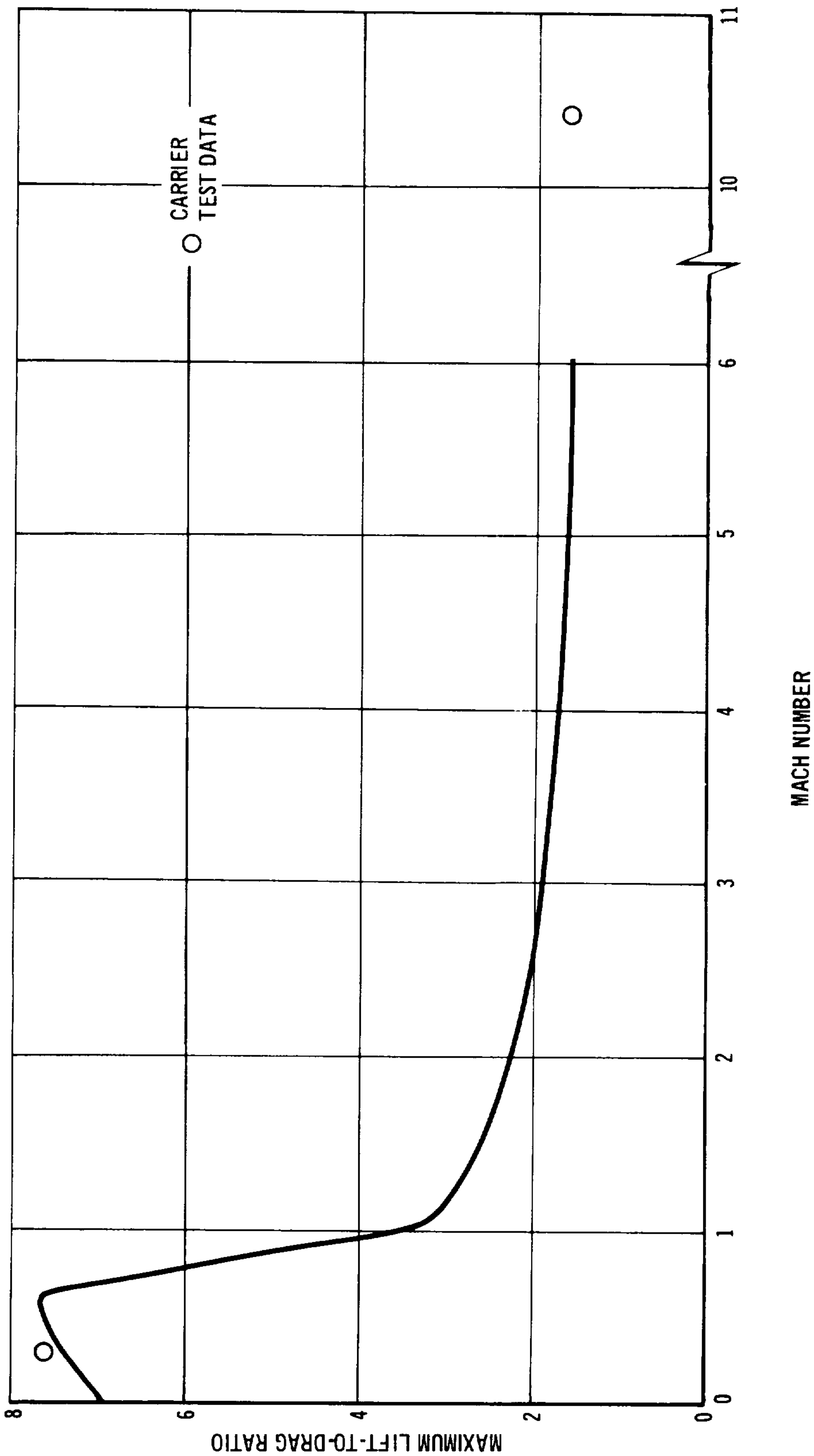


Figure 3-55

CARRIER CONFIGURATION
VARIATION OF ELEVON LIFT EFFECTIVENESS
WITH MACH NUMBER
ZERO ANGLE OF ATTACK

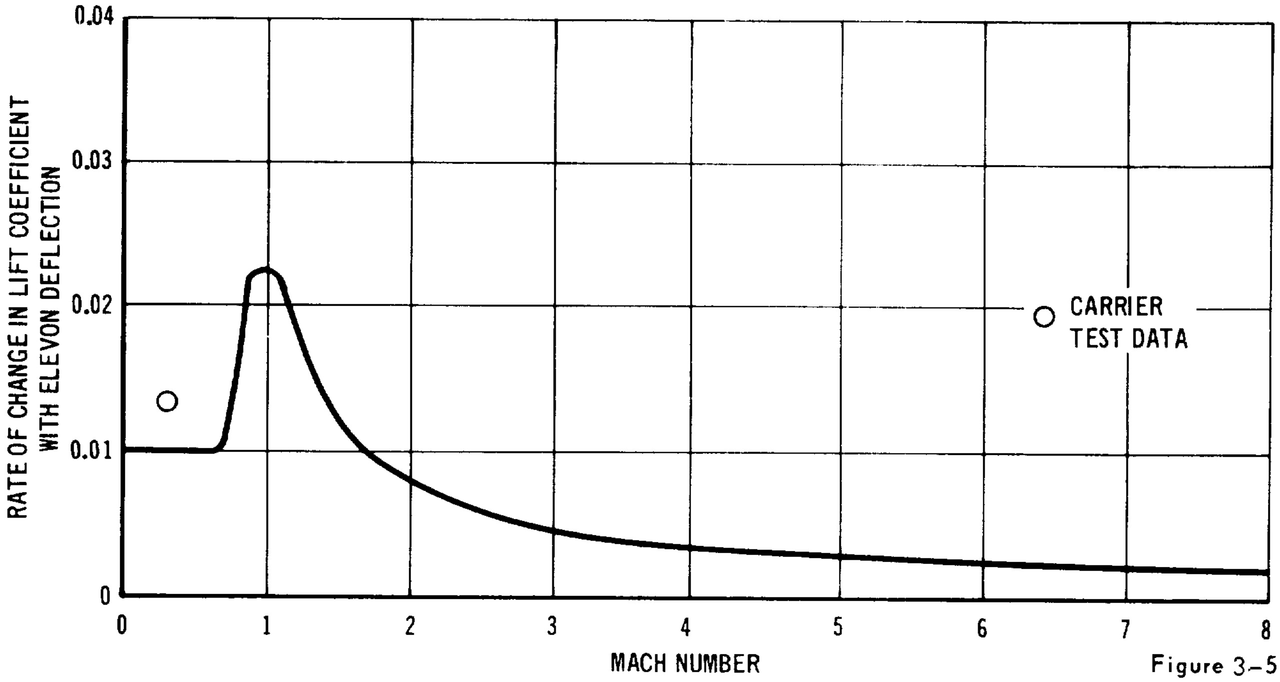


Figure 3-56

CARRIER CONFIGURATION
VARIATION OF ELEVON MOMENT EFFECTIVENESS
WITH MACH NUMBER
ZERO ANGLE OF ATTACK

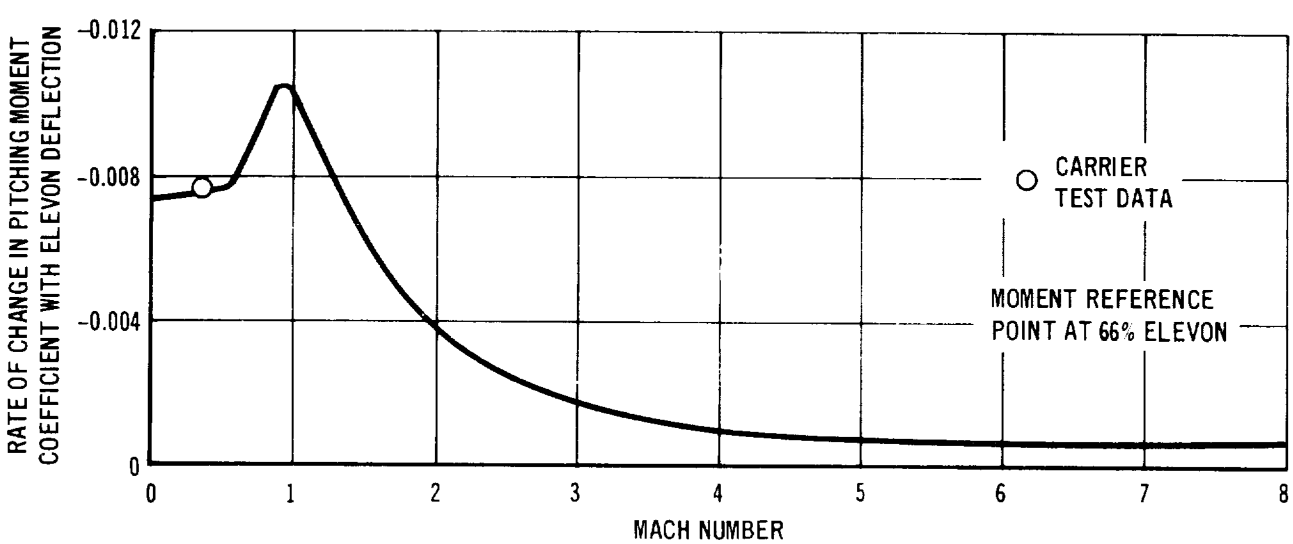


Figure 3-57

CARRIER CONFIGURATION
VARIATION OF SIDE FORCE DERIVATIVE WITH MACH NUMBER
ZERO ELEVON DEFLECTION

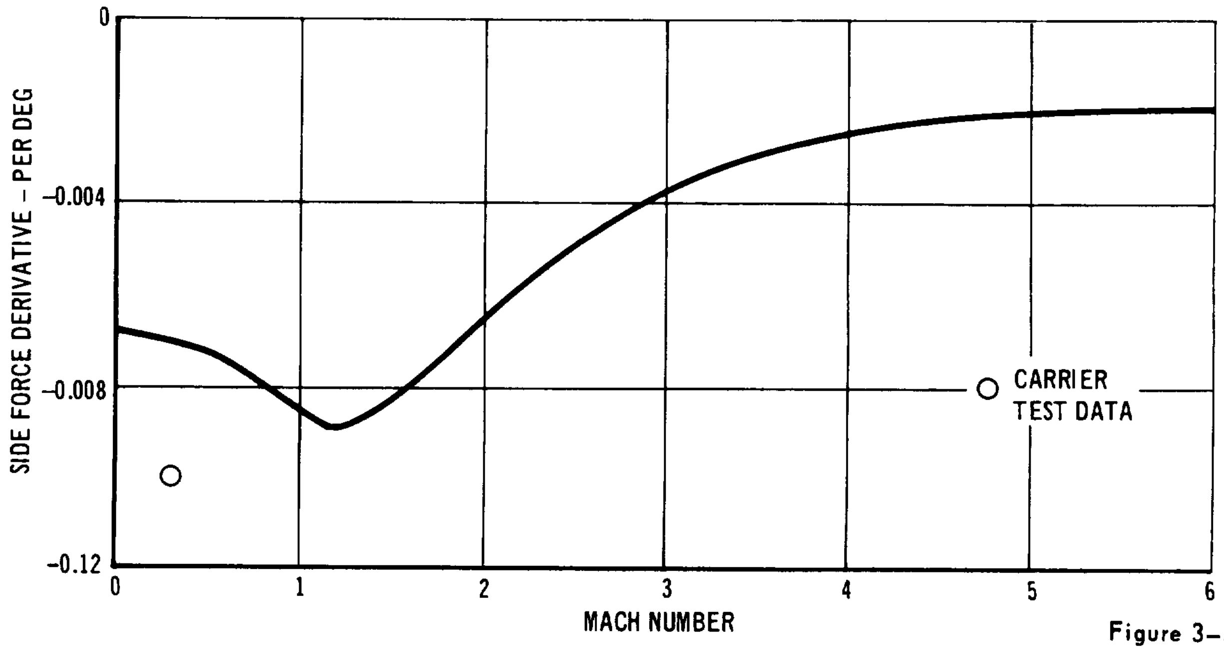


Figure 3-58

CARRIER CONFIGURATION
VARIATION OF YAWING MOMENT DERIVATIVE WITH MACH NUMBER
ZERO ELEVON DEFLECTION

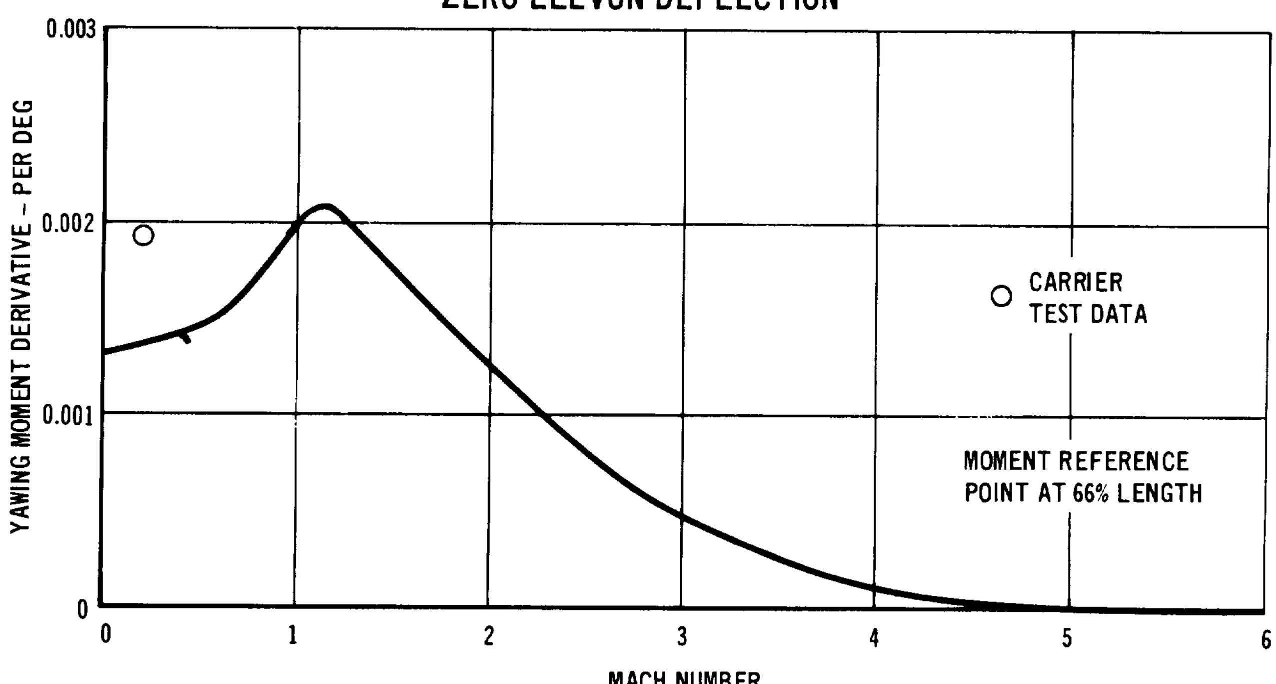


Figure 3-59

CARRIER CONFIGURATION
VARIATION OF ROLLING MOMENT DERIVATIVE
WITH MACH NUMBER
ZERO ELEVON DEFLECTION

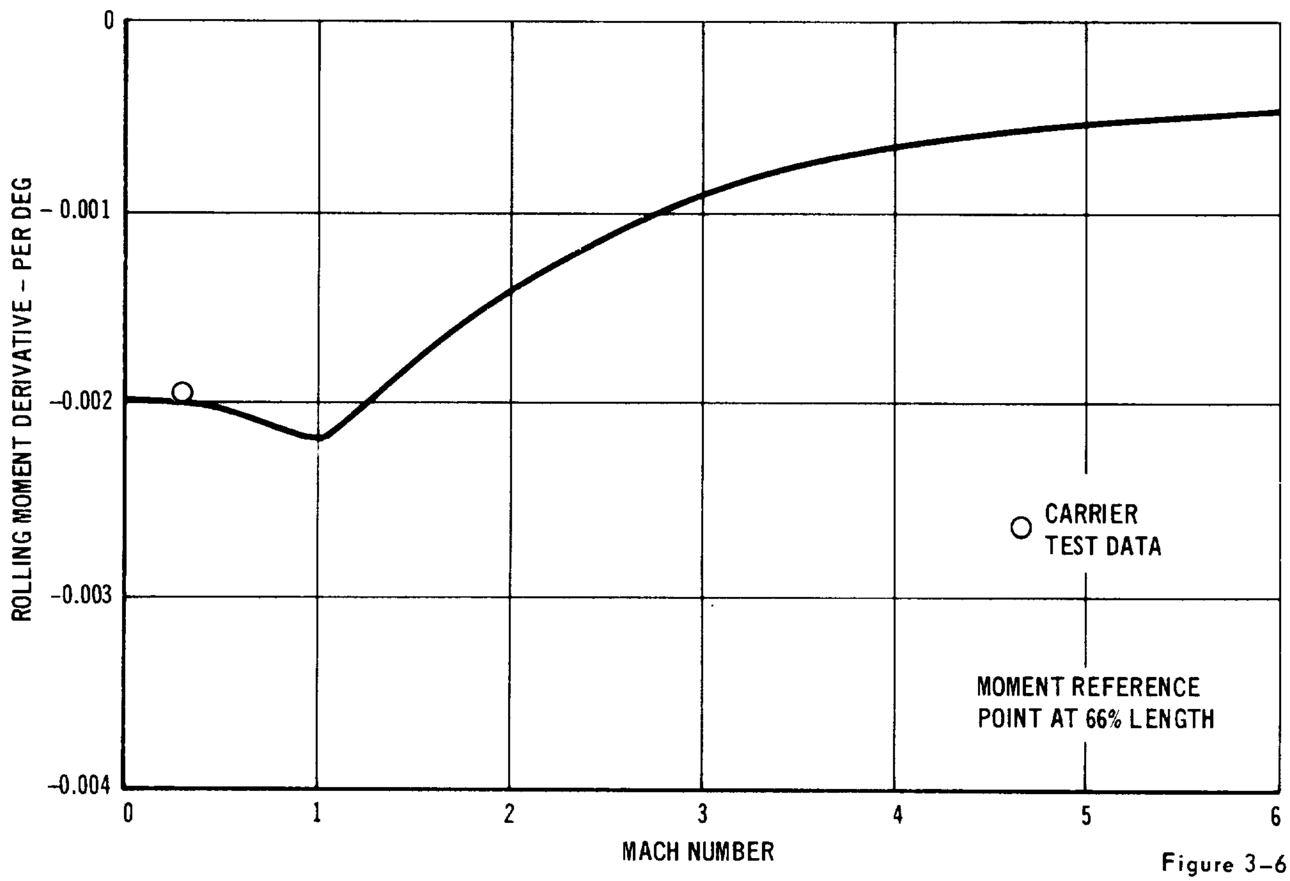


Figure 3-60

b) Carrier Entry Trajectory Analysis - The concept of reusability requires that the carrier be recovered at a remote site or fly back to the original launch site. As noted in the design evolution studies in Volume I, the payload geometry effectively sizes the Orbiter length (107 ft) and consequently the boost ΔV distribution between the two stages. In the earlier version of the orbiter, non integral boost propellant tanks were employed and the total ΔV which could be uncorporated in the orbiter was considerably less than what was later found possible if integral propellant tanks were employed. With the non-integral orbiter tanks, first stage separation occurs at high velocity and at a significant distance from the launch site. For this case carrier landing at a remote site appeared attractive. However, by going to the integral tank concept for the orbiter, a relatively lower staging velocity was achieved and the requirements for returning the carrier to the original launch site became much less severe. Re-entry trajectories permitting carrier return to the launch site were therefore further analyzed.

The objectives of the carrier reentry trajectory shaping were to minimize the down range flight within thermostructural constraints, thus minimizing the weight penalty resulting from thermo protection subsystem design and the cruiseback fuel requirements.

The principal variables in this study were separation conditions (velocity, altitude, and flight path angle), angle of attack, bank angle, and area loading. Two approaches were available for solution of this problem, systematic parametric studies or optimization techniques employing the calculus of variations or steepest descent. The latter had the advantage of yielding the true optimum but the disadvantages of providing less insight to the problem as well as requiring a longer development period. Consequently, the parametric approach was chosen. The results of these studies are discussed in the subsequent paragraphs.

High Angle of Attack - The initial parametric study addressed the feasibility of pullout without violation of the temperature constraint. Primary variables were separation velocity and angle of attack. The effects of separation velocity on pullout environment were studied by selecting separation conditions from representative launch trajectories. Velocities ranged from 7,253 to 16,518 ft/sec with corresponding flight-path angles from 15.0 to 2.9 degrees and altitudes from 215,381 to 347,667 feet. From each separation condition, three trajectories were calculated. Each had a normal load factor limit of 3 g's but with angle of attack limits of 30, 40, and 50 degrees. Each trajectory was flown in an unbanked attitude to a horizontal flight path and terminated at that point because the critical environment had been encountered. For the 50 degree angle of attack cases, the resulting pullout dynamic pressure was about 200 lbs/ft^2 except near separation velocities of 9,000 ft/sec where it was over 300 lbs/ft^2. It should be noted that the dynamic pressures mentioned above are significantly higher than the final mission trajectory values because the reference launch trajectories went to a 66 x 100 na. mi. orbit instead of the current 45 x 100. The lower angle of attack limits yielded proportionately higher dynamic pressures. At pullout, the load factor constraint kept angles of attack below their limiting values. Thermodynamic analyses indicated the higher angle of attack limits and slower speeds yielded lower peak temperatures. Thus, it was concluded that high angle of attack pullout maneuvers could be performed without violating temperature or load factor constraints.

Bank Angle - Bank angle was introduced into the reentry maneuver commands to reduce downrange distance and simultaneously turn the carrier back toward the launch site. Unlike the thermal environment criterion which was a yes or no situation, the problem of range reduction was one of degree. Primary variables for this study were initial conditions of velocity, altitude, and flight-path angle and were the same conditions as used in the previously discussed study.

The angle of attack limit was 50 degrees and the carrier was banked to 45 degrees after pullout. The trajectories were flown to a speed of 1000 ft/sec which for all practical purposes expended the unpowered range making capability. Also, at such low speeds, heading angle is readily changed and it was assumed that the vehicle was pointed back towards the launch site at initiation of the cruise. The following table gives the resulting range from the launch site as a function of the separation velocity for the 45 degree banked trajectories.

Velocity	7,253	9,142	13,013	16,518	ft/sec
Range	320	510	880	1,520	na. mi.

Because other variables were not optimized, these ranges were excessively high. They did, however, indicate the velocity-range relationship.

The study of the effects of bank angle was expanded. The objective was to determine the sensitivity of cruiseback range to bank angle for different angles of attack. Three angles of attack were considered (30, 40 and 50 degrees) and two bank angles were considered in this parametric. However, only one set of initial conditions were considered; i.e., 13,013 ft/sec, 284,443 feet and 6 degrees flight-path angle.

From these separation conditions the flight commands specified a 3g pullout maneuver with angle of attack limits of 30, 40, and 50 degrees. Each trajectory was flown initially in an unbanked attitude to a level flight condition and then banked to the respective angle, i.e., 45 or 60 degrees. During the lateral maneuver segment, bank angle was programmed to a constant value and angle of attack was selected to minimize altitude oscillations.

The range sensitivity to angle of attack was between 3 and 3.5 na. mi. per degree whereas the sensitivity to bank angle was of the order of 9 na. mi. per degree and these sensitivities are independent. Thus, maximum range reductions are achieved with combinations of high angle of attack and high bank angles.

The high drag attendant with high angle of attack maneuvers resulted in significant deceleration at high altitude such that the altitude-velocity profile of these trajectories did not violate the temperature boundary. At a given velocity the altitude margin between the curves was a measure of the additional manuever-ability available because increased bank angle reduces the vertical lift component and subsequently the flight altitude.

Inverted Flight Segment - Parallel studies of the ascent phase trajectory shaping resulted in the recommendation for lower altitude orbit insertion. For the previously discussed carrier reentry studies, initial conditions corresponded to trajectories shaped for insertion into a 66 x 100 na. mi. orbit. The revised insertion conditions were for 45 x 100 na. mi. orbits. The reshaped ascent-phase trajectories resulted in lower separation altitude and shallower flight-path angle. These effects enhanced the reduction of downrange travel after separation. The lower altitude produced a higher-dynamic pressure environment which resulted in additional deceleration. The shallower flight-path angle reduced apogee altitude.

Additional reduction in apogee altitude was achieved by flying inverted at high angle of attack beyond apogee to a predetermined negative flight-path angle followed by roll out to a wings level attitude for the pullout.

The following variables were considered:

1) Initial conditions
2) The negative flight-path angle beyond apogee when inverted flight was terminated.
3) The bank angle after pullout.

Angle of attack was programmed to 50 degrees throughout the reentry maneuver. Inverted flight was to -1, -3, and -5 degrees. Bank angles after pullout were 60, 70 and 80 degrees. Separation velocities considered were 8,171, 9,997, 11,856 and 13,719 ft/sec. The results from this study were:

1) Range reductions on the order of 10 to 20 na. mi. for flying inverted to -3 degrees rather than to -1 degree is possible, but angles of -5 degrees give additional improvements of no more than 3 na. mi.
2) Range reductions of 10 to 30 na. mi. for increasing bank angle from 60 to 70 degrees were possible (except for the lowest separation speed case).
3) Increasing bank angle to 80 degrees further reduced the range, but there is a "drop off" point near 80 degrees where the vehicle sink rate is too high and the dynamic pressure peak increases by a factor of 10.

It was concluded that inverted flight to a -3 degree flight path angle was advantageous, and a bank angle of approximately 70 degrees was nearly optimum for the problem formulated and the constraints imposed.

<u>Nominal Missions</u> - The flight profile derived from the parametric studies culminated in definition of the final nominal mission trajectory definition. Initial conditions correspond to the nominal ascent trajectory presented in Section 3.1.1. The altitude-range profile and ground track are shown in Figure 3-61 and shows a cruiseback range of 357 na. mi. Time histories of significant trajectory parameters are presented in Figures 3-62 and 3-63. Of special note are the moderate dynamic pressure and load factor.

<u>Summary</u> - Systematic investigation of significant variables resulted in a flight profile which minimized return range within specified heating and load constraints. Cruiseback to the launch site can be incorporated into the design of a two-stage fully reusable space transportation system.

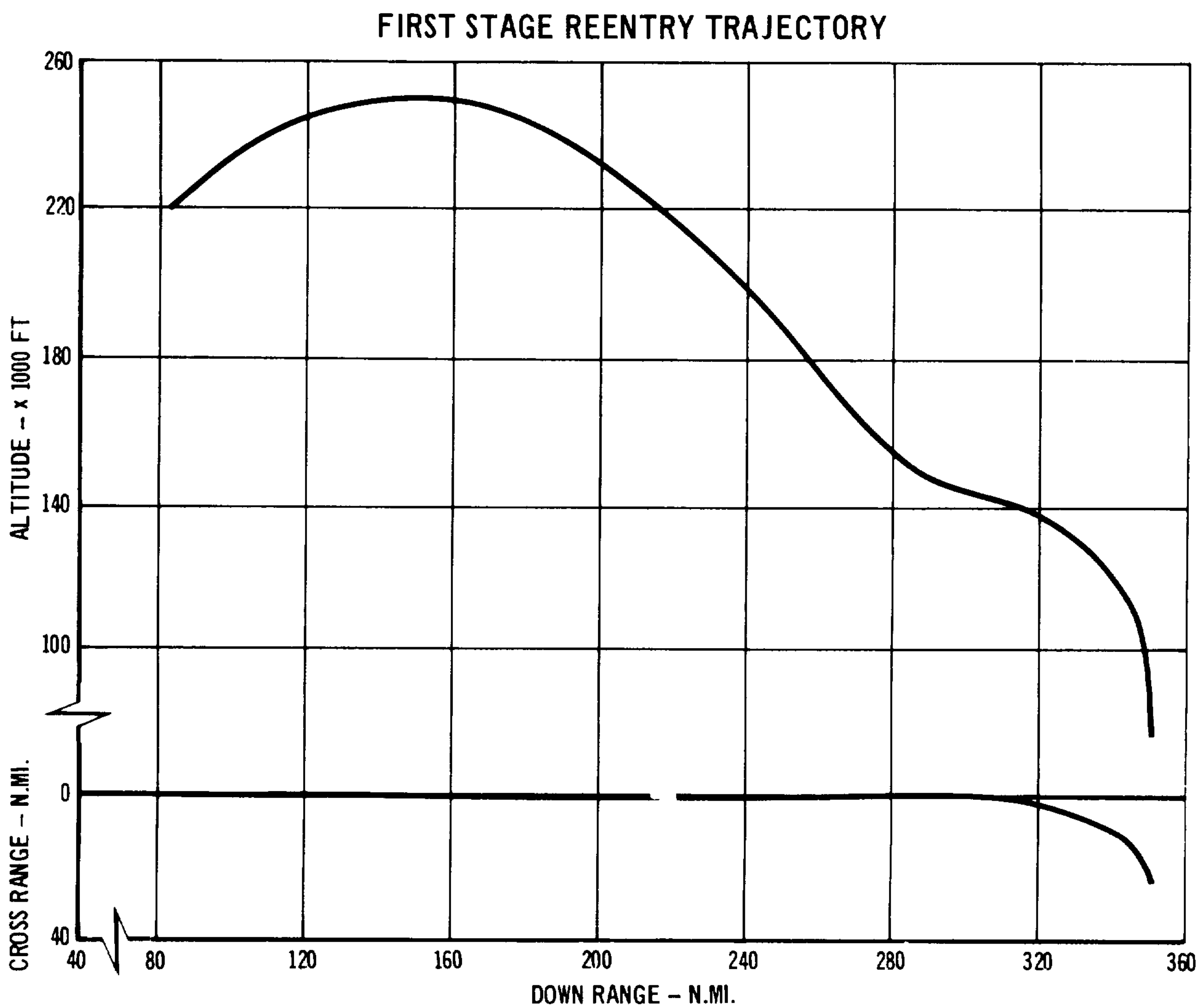

FIRST STAGE REENTRY TRAJECTORY
260
220
180
140
100
0
40
ALTITUDE – x 1000 FT
CROSS RANGE – N.MI.
40
80
120
160
200
240
280
320
360
DOWN RANGE – N.MI.

Figure 3-61

FIRST STAGE REENTRY TRAJECTORY PARAMETERS
Part II

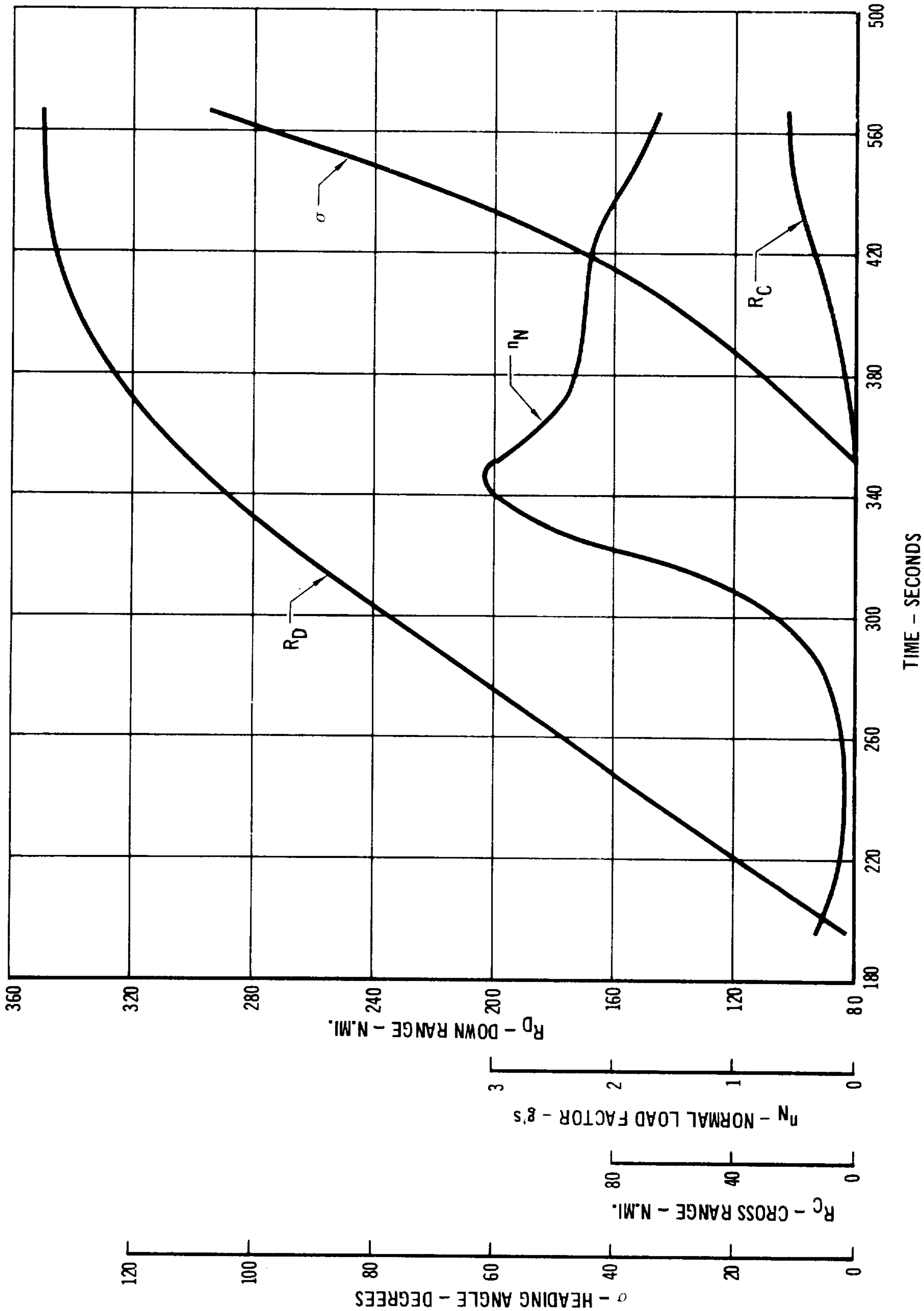


Figure 3-62

FIRST STAGE REENTRY TRAJECTORY PARAMETERS
Part I

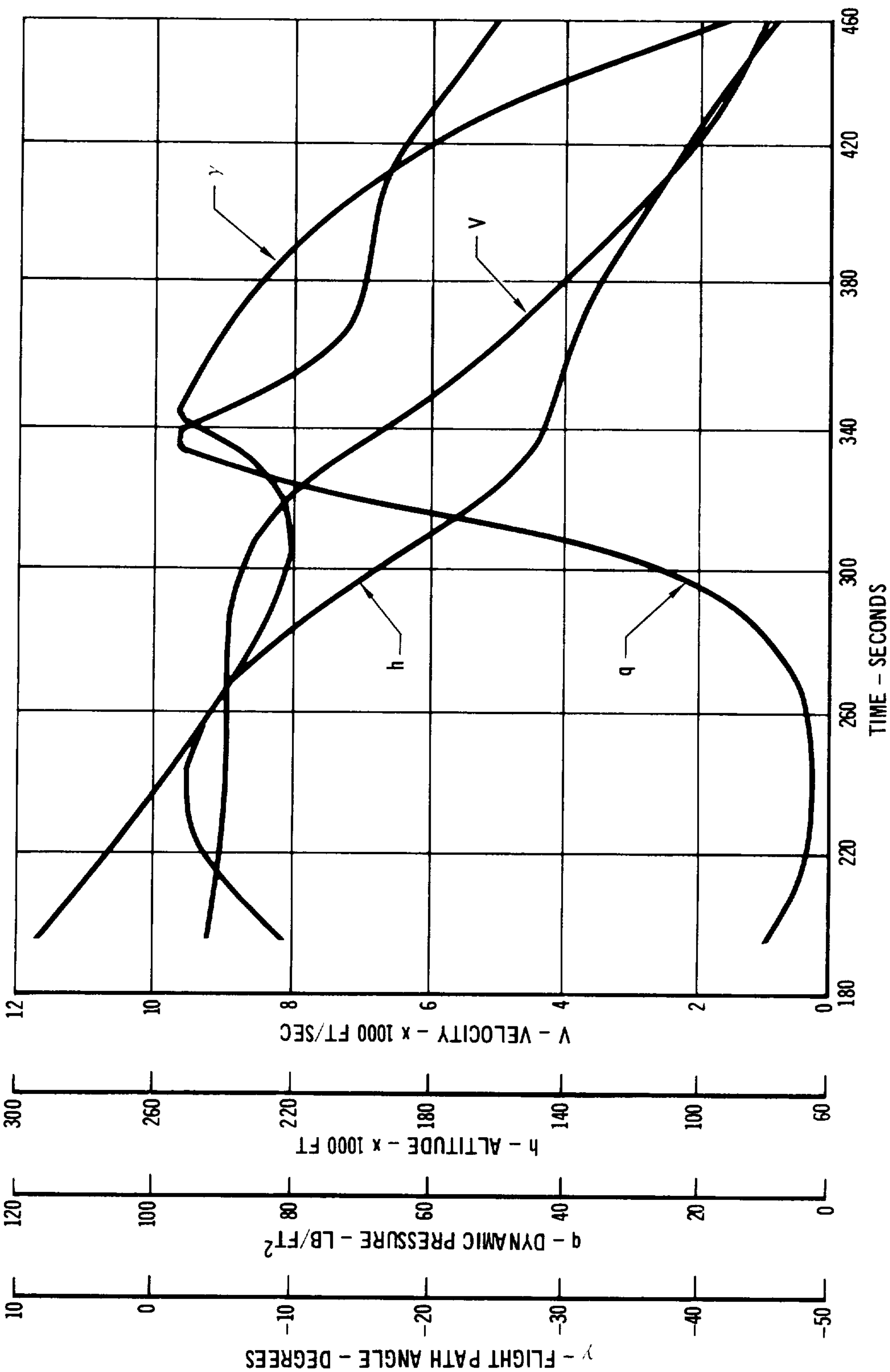


Figure 3-63

c) Carrier Entry Heating Analysis - Maximum temperatures experienced by the Carrier during a nominal launch and reentry for an impulsive velocity of 14,500 ft/sec (actual ΔV is 9170 ft/sec) are shown in Figure 3-64.

These temperatures are for laminar flow and, except for the leading edge of the dorsal fin, are based on heat transfer test data obtained from NASA-LRC. Experimental heat transfer tests were conducted on both high and low wing clipped delta configurations at a Mach number of 10.4 and a Reynolds number of 0.5×10^6 based on model length. The model was coated with a phase change material and local heating rates were determined by interpretation of photographic data. Lines of constant heating rates as interpreted from the photographic data are shown in Figures 3-65 and 3-66, Values shown are ratios of local heating rates to a calculated stagnation point heating rate on a hemisphere having a diameter equal to the vertical thickness of the model. The dorsal fin is shielded at an angle of attack of 50° during reentry. Consequently, temperatures for the dorsal fin leading edge were determined from swept cylinder theory for ascent flight conditions at an angle of attack of zero degrees. Although the leading edge radius decreases with distance from the base of the dorsal fin, estimated temperatures near the base are higher because of allowance for bow shock wave impingement.

As shown in Figure 3-66, temperatures for laminar flow along the bottom surface are in the range of 800 to 900°F. However, peak heating during reentry occurs at relatively low altitudes (less than 160,000 ft.) and the flow will be turbulent based on the criterion of onset at a local Reynolds number of 10^6 and fully developed turbulent flow at 2.0×10^6.

In order to investigate the influence of turbulent flow on maximum temperatures in the absence of test data, blunt modified Newtonian flow was assumed to define local flow properties. It was also assumed that streamline divergence or outflow has little influence on turbulent heating rated and equilibrium temperatures. Based on these assumptions, the variations of local Reynolds number and turbulent temperatures for a wetted length of 50 feet on the lower surface centerline are illustrated for Carrier reentry flight conditions in Figure 3-67. Consequently, assuming that transition occurs at a local Reynolds number of 1.0×10^6 and that fully-developed turbulent flow exists at a Reynolds number twice the value at transition onset, it is seen that maximum temperatures along the bottom surface would be 1150°F which is 350 degrees higher than the maximum laminar temperatures of approximately 800°F shown on Figure 3-66. With onset and fully developed turbulent flow occurring at 0.5×10^6 and 10^6, respectively, maximum temperatures would be approximately the same.

MAXIMUM CARRIER TEMPERATURES

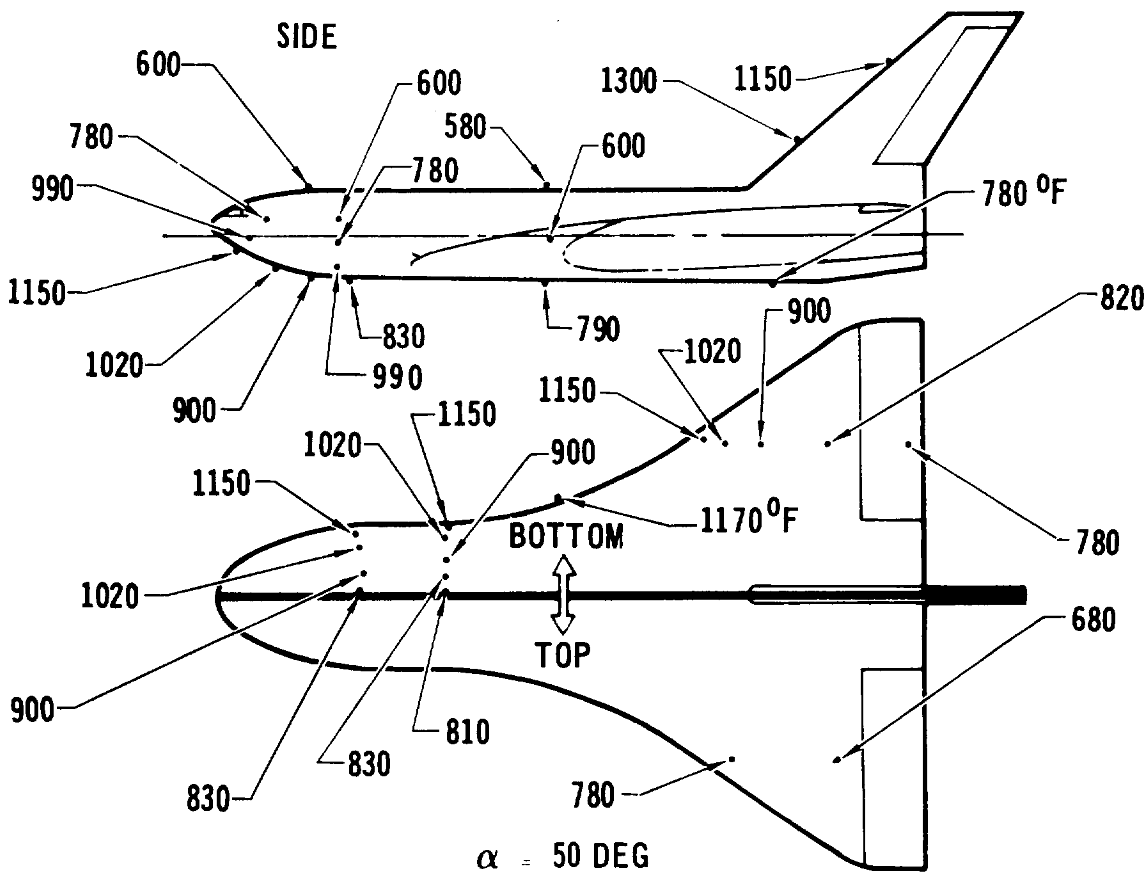


Figure 3-64

CARRIER EXPERIMENTAL HEATING

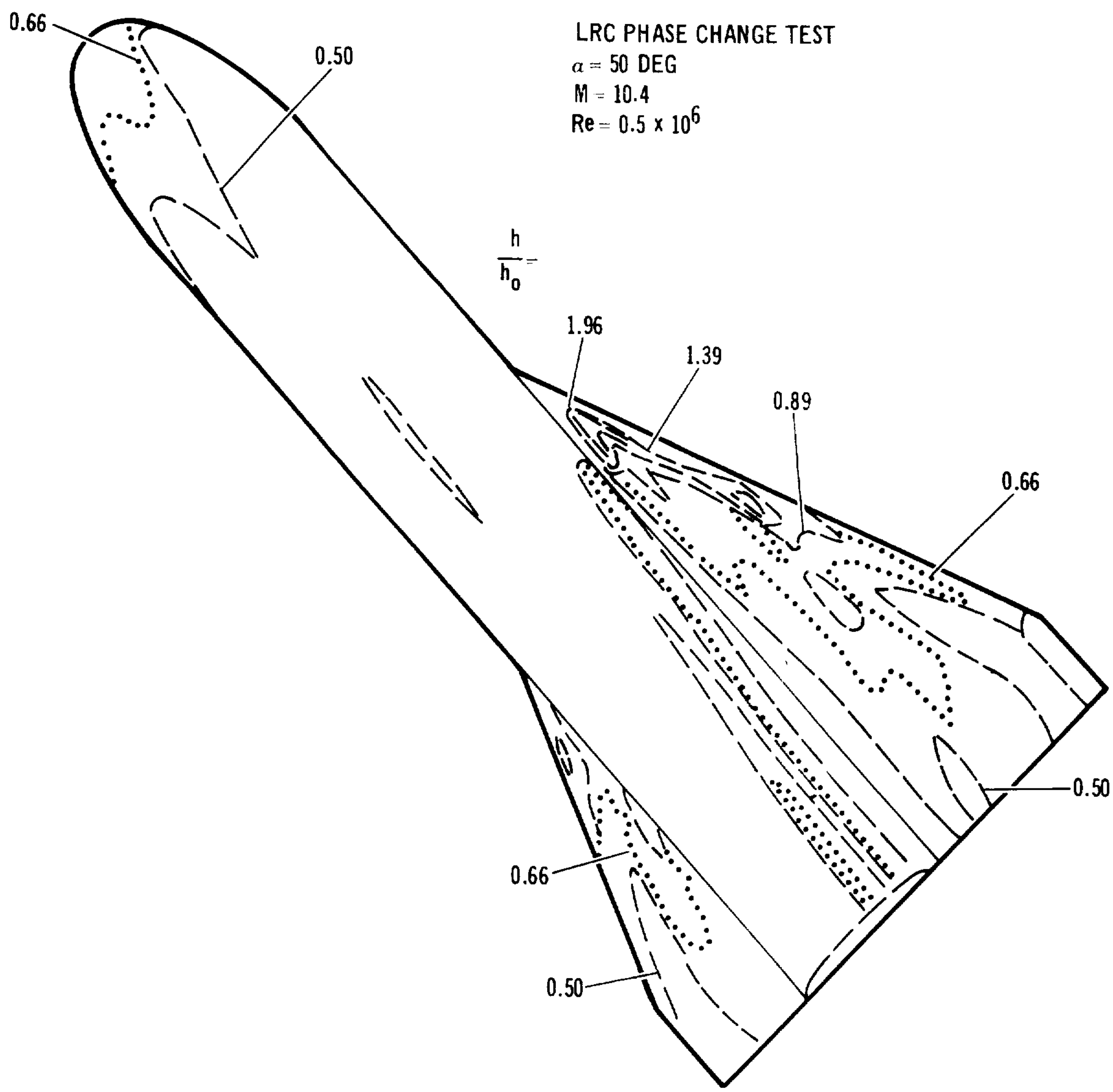


Figure 3-65

CARRIER EXPERIMENTAL HEATING

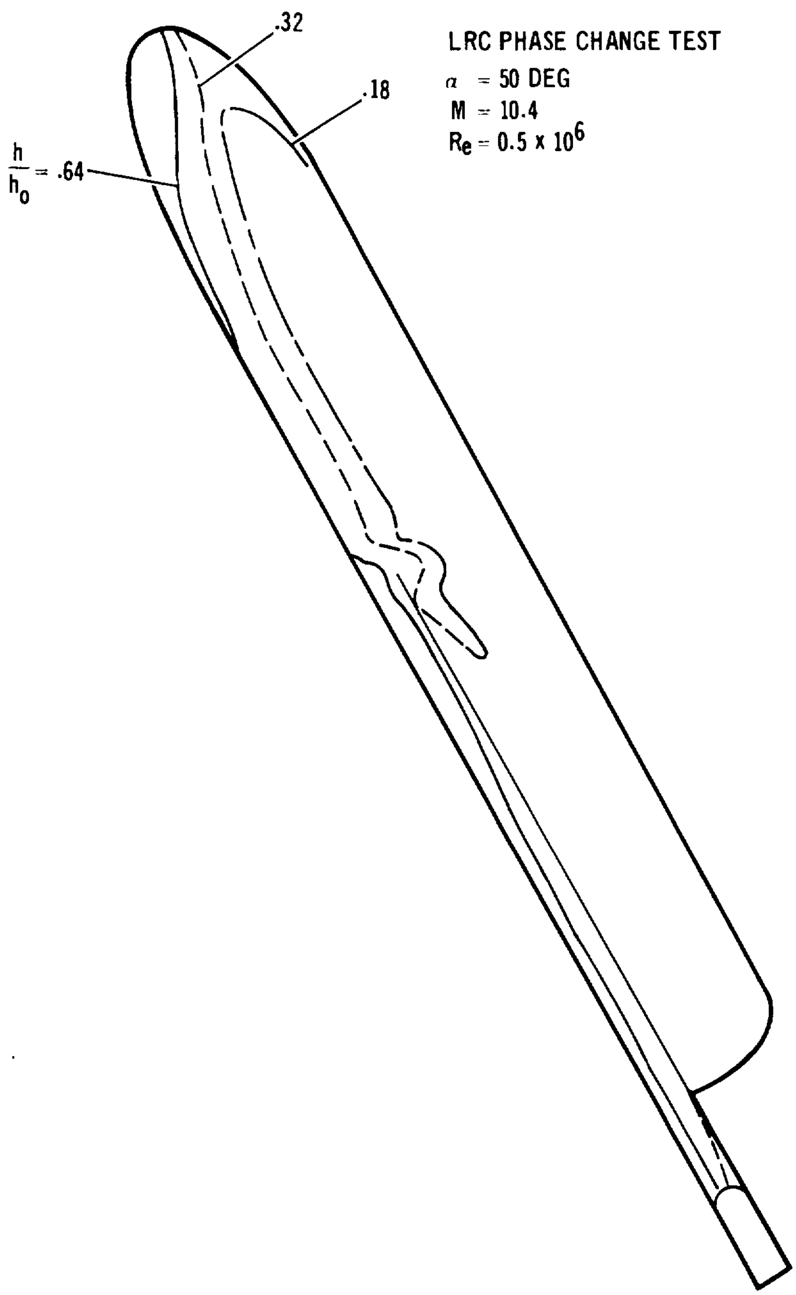


Figure 3–66

CARRIER TURBULENT HEATING EFFECTS

- MODIFIED NEWTONIAN FLOW

- δ = 50 DEG
- S = 50 FT
- REF ENTHALPY
- STRIP THEORY

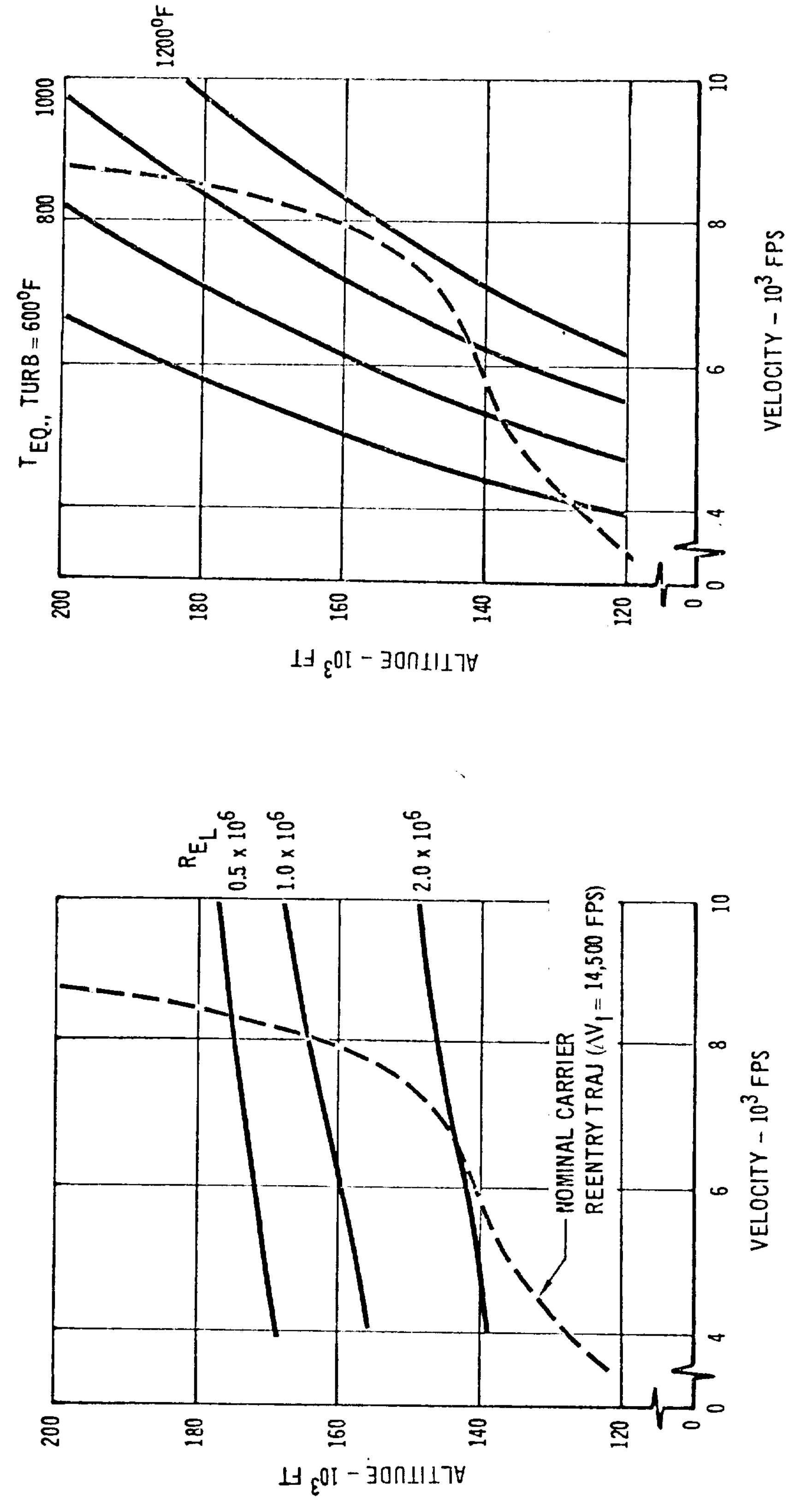


Figure 3-67

3.3.2 Orbiter Entry Performance - The entry performance analyses for the orbiter are presented in this section. The aerodynamic performance is discussed first followed by the trajectory and heating analyses.

a) Orbiter Entry Aerodynamic Analysis - Orbiter models of various sizes have been tested in various facilities, accumulating almost 10,000 hours of wind tunnel testing in about seven and a half years of research and development. Models as small as 4.5 inches and as large as 28 feet have been tested up and down the Mach number range, and in more recent years, flight test results from this configuration have veen obtained through the transonic Mach number range. A large portion of these test results have been published in some 30 classified documents by the NASA. These results have been used in the various facets of this study. However, all of the data could not be used directly. There were areas where it was found necessary to revise the published information. For example, the trimmed test data was presented with a moment reference point at the 53% station on the vehicle. But as the orbiter design began to "take shape" it was apparent that the center of gravity of the vehicle would be at the 54% station, over a large portion of the flight envelope. Thus, it becomes necessary to reexamine the orbiter trim characteristics to determine the effects of trimming with a farther aft center of gravity. This investigation indicated that the vehicle trim characteristics changed only slightly. Furthermore, the vehicle directional stability still remained adequate.

Since the transonic region is the most critical area for the directional stability of the orbiter configuration, it was felt that the aft center of gravity location may cause the vehicle to become unstable at these flight conditions. However, examination of the data transferred to the new moment reference point indicated that the vehicle would remain stable at least up to 28° angle of attack.

b) Orbiter Entry Trajectory Analysis - Entry trajectory shaping was examined parametrically to provide an insight into the interaction of the available control variables with the competing performance requirements. Parametrics were employed in pullout, equilibrium glide, and the transition from equilibrium glide to subsonic flight. The following table summarizes variables investigated:

Phase	Variables
Pullout	Angle of Attach
	Load Factor
	Reentry Angle
	Reentry Velocity
	Weight
Glide	Angle of Attack
	Bank Angle
Transition	Pushover Speed
	Fall Distance
	Bank Angle
	Load Factor

In addition to parametric studies, two specific reentry trajectories were calculated. The baseline reentry was flown to reach 390 nautical miles of cross range while keeping total flying time as small as possible. The 390 nautical miles guarantees return capability once each 24 hour period. A twice per day return trajectory which allowed reentry time to vary while peak index temperature was held constant at 2200°F, was also analyzed. These trajectories were derived primarily for purpose of thermal protection system evaluation which is discussed in part C of this section and section 4.2 of Volume I.

Pullout - The pullout maneuver is executed during that portion of flight between reentry (400,000 feet) and the point in the trajectory when the flight-path angle is approximately zero, e.g., -0.12 degrees. One of the primary objectives of the pullout maneuver is to minimize the peak temperature which generally occurs at the pullout point. Since for a given set of reentry conditions the pullout velocity is nearly independent of the maneuver and since temperature decreases with increasing altitude and increases with increasing angle of attack, the objective is to pull out at high altitude and low angle of attack. These two conditions oppose each other since a decrease in angle of attack decreases the lift and lowers pullout altitude. Therefore, a trade-off between pullout angle of attack and pullout altitude was in order. One way to make this trade-off was to fly at maximum angle of attack until a specified load factor was reached and then maintain that load factor to the pullout point. A better method was to fly at high angle of attack to some parametrically determined temperature and then modulate to maintain a load factor that gives the same temperature at pullout. Both of these methods have been tried with some success. However, improvement over constant angle of attack pullouts was small. This, together with the fact that equally high temperatures occurred on the equilibrium glide path prompted the selection of unbanked, constant angle of attack pullouts.

The effect of reentry conditions on the pullout point was studied by parametrically varying reentry velocity, flight path angle, and weight. The following table summarizes the sensitivities of pullout parameters to reentry parameters.

Pullout Variable, P	Reentry Variable, r	dp/dr
Velocity	velocity	$1.35 \frac{ft/sec}{ft/sec}$
Altitude	velocity	$25 \frac{ft}{ft/sec}$
Velocity	flight path angle	$450 \frac{ft/sec}{degree}$
Altitude	flight path angle	$27,000 \frac{ft}{degree}$
Velocity	wing loading (w/s)	$1.7 \frac{ft/sec}{lb/ft^2}$
Altitude	wing loading (w/s)	$-500 \frac{ft}{lb/ft^2}$

Note that .2 degrees of flight-path angle or 200 ft/sec of velocity is equivalent to 10 lb/ft^2 in wing loading as far as pullout altitude is concerned. Also, wing loading and reentry angle have quite small effects on pullout velocity. The significantly greater than one-for-one return on velocity is also noteworthy.

Equilibrium Glide - Equilibrium glide is defined as that portion of the reentry from the pullout point to a velocity of about 4000 ft/sec where a pitch over maneuver is initiated.

For a given vehicle and a specified pullout, two variables, angle of attack and bank angle determine the equilibrium glide. This in turn affects down range, cross range, heating time, peak temperature, peak load factor, etc. Therefore, considerable analysis was done to select these two control variables.

Equilibrium glide exists when weight is balanced by the sum of vertical lift and centrifugal relief. Consequently for a given bank angle and velocity the angle of attack determines the altitude and hence the temperature. With angle of attack chosen to yield either a large cross range or a short time, bank angle was selected to satisfy constraints in load factor, temperature, and dynamic pressure. Within the limits of these constraints bank angle was the important variable in maximizing cross range or minimizing time.

The classic method of flying glides is to maintain constant altitude after pullout until a predetermined bank angle is reached and then continue at that bank angle. This is a very good method because it gives large cross ranges at moderate peak temperatures and reasonable flight times. It can, however, be improved on in several ways depending on the mission requirements.

For example, it was found that cross range could be increased without a peak temperature penalty by a special means of bank angle modulation. If on a plot of altitude vs velocity, one draws a temperature boundary and a constant bank angle glide track they are tangent at about 20,000 ft/sec. This means at all other velocities there is a bank angle maneuvering margin. High bank angles above 20,000 ft/sec have been found significant in increasing cross range. The constrained maximum cross range trajectory discussed below demonstrates this by providing 586 nautical miles of cross range with the same peak temperature attained by unbanked flight.

Transition - Transition is that porttion of flight between the end of glide and the beginning of the subsonic approach and landing. The transition maneuver was executed by performing a zero-lift descent to an altitude, h, followed by a constant load factor, n_N, through pullup. Initial velocity, h, and n_N were parametrically adjusted. The "high key" conditions necessary for approach were not quite met as a result of this first parametric. All trajectories flown in this manner were too fast at the desired altitude. To correct this, a bank angle during the pullup was used. The resultant trajectory is described below as the end of the baseline reentry.

Baseline (Once/Day Return) Reentry - The baseline reentry is defined as having the minimum possible time subject to a 2200°F temperature constraint aft of the 12.5% reference point, a 3g normal load factor constraint, and a 390 nautical mile cross range constraint which insures once/day return capability. This trajectory was derived for purposes of evaluating the baseline thermal protection concept which uses TDN_iC_r. (See section 4.2 of Volume I). It was executed as follows: At reentry the vehicle was rolled to 65.8 degrees and pitched to 50 degrees. This attitude was held to a velocity of 24,000 ft/sec where the peak temperature was reached. From there bank angle was modulated to minimize the lift-to-drag ratio while not violating the constraints on temperature and load factor. Between velocities of 10,000 and 6,000 ft/sec the bank angle was reduced to zero in order to stretch out the range and satisfy the 390 nautical miles cross range constraint. The transition to subsonic flight was performed by a ballistic descent beginning at 4,000 ft/sec and ending at an altitude of about 50,000 ft. Below 50,000 ft a 3g pullup with a 50 degree bank angle was executed to satisfy the "high key" conditions.

Figure 3-68 presents altitude, down range, and cross range for this trajectory. Significant time histories are given in Figures 3-69 through 3-71. The altitude velocity profile is compared to the twice/day return reentry in Figure 3-72.

Twice/Day Return Reentry - This trajectory also satisfies the 2200° and 3g maximum temperature and normal load factor constraints while providing 586 miles of cross range which insures a twice/day return capability. This trajectory has been calculated and was executed as follows: At reentry the vehicle was pitched to 30 degrees angle of attack and flown unbanked to pullout. Bank angle modulation was begun by rolling 89.6 degrees. Then while modulating angle of attack between 26 and 28 degrees to damp oscillations, bank angle was gradually reduced until it was zero at 20,200 ft/sec. The velocity was chosen as being critical in that it produced the peak temperature for unbanked flight. Below 20,200 ft/sec the maneuvering margin increased and the bank angle was gradually increased to 77 degrees at 11,905 ft/sec. By this point the heading angle had been changed 28 degrees which made it advantageous to begin increasing vertical lift-to-drag ratio (L/D) so that range could be stretched out. This was done at the expense of increased heading change by slowly reducing bank angle to zero at 8000 ft/sec. From 8000 to 3900 ft/sec straight ahead flight at near (L/D) maximum was performed. The transition to subsonic flight was then begun with a ballistic fall to 45,000 ft. To pull up and reach high key conditions the vehicle was then banked 50 degrees while angle of attack was modulated to produce a normal load factor of 2.7 g's. Figure 3-73 presents altitude, down range and cross range for this trajectory. Significant time histories are given in Figures 3-74 through 3-76.

SECOND STAGE BASELINE REENTRY

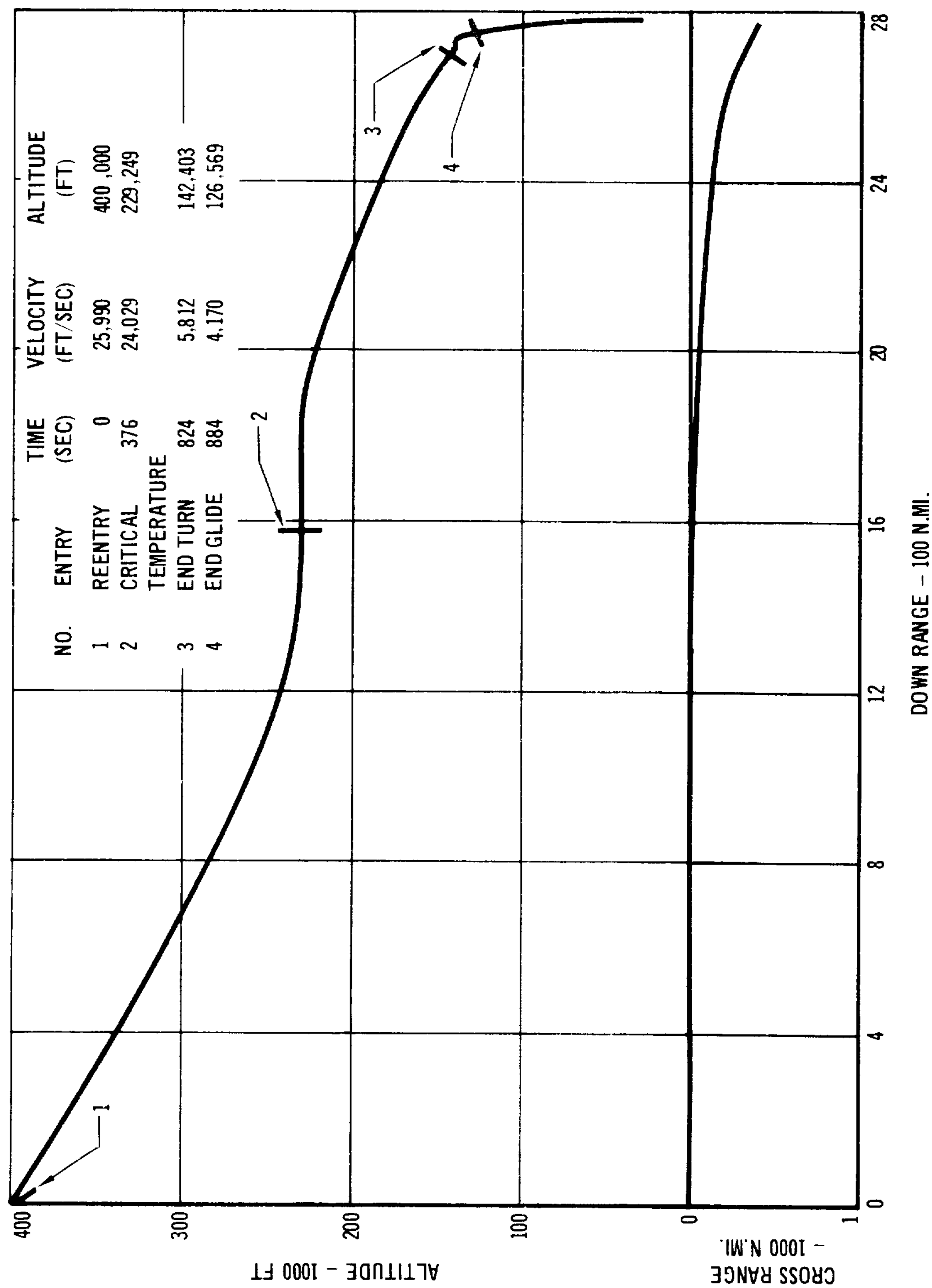


Figure 3-68

SECOND STAGE BASELINE REENTRY TRAJECTORY PARAMETERS
Part I

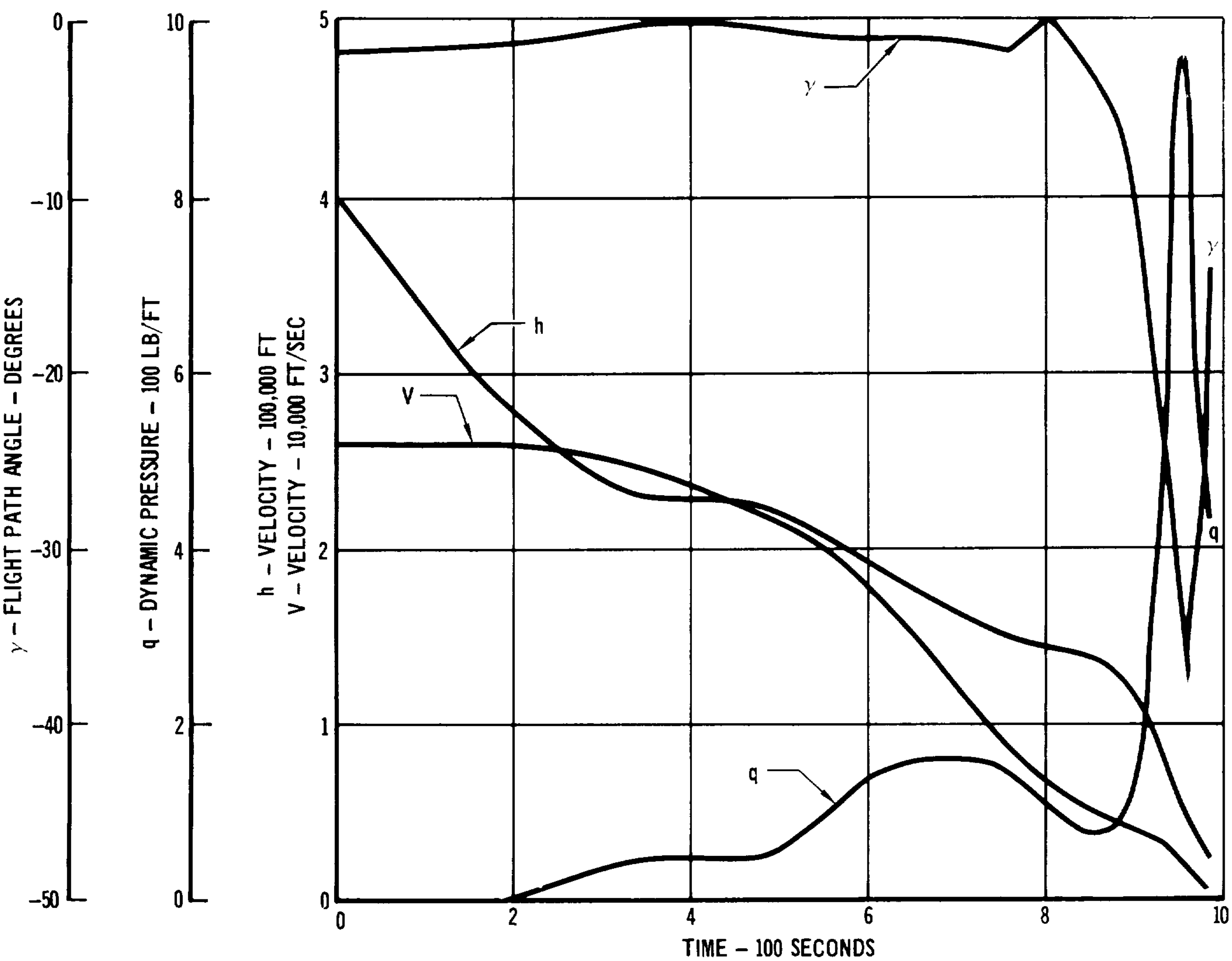


Figure 3-69

SECOND STAGE BASELINE REENTRY TRAJECTORY PARAMETERS
Part II

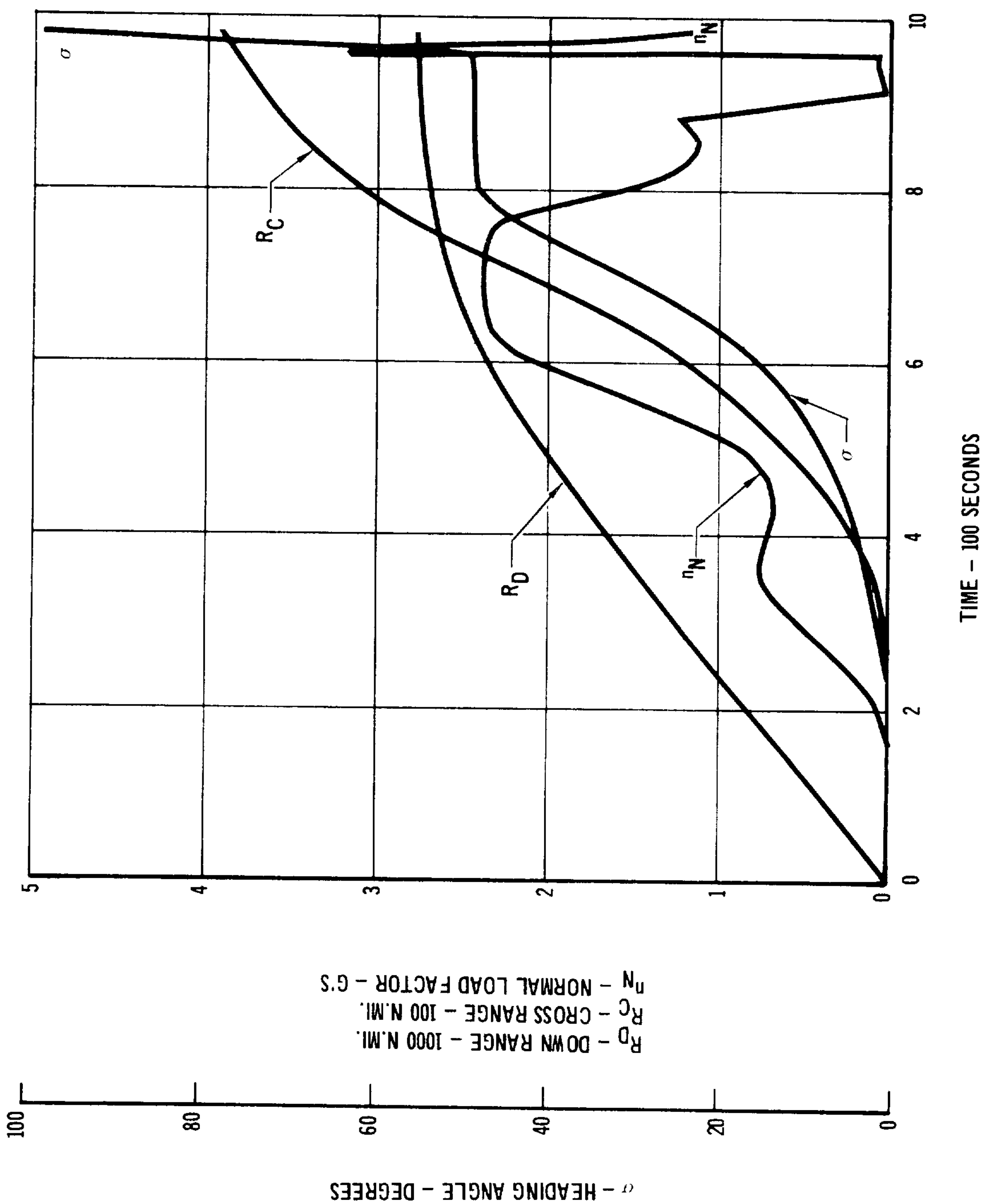


Figure 3-70

SECOND STAGE BASELINE REENTRY TRAJECTORY PARAMETERS
Part III

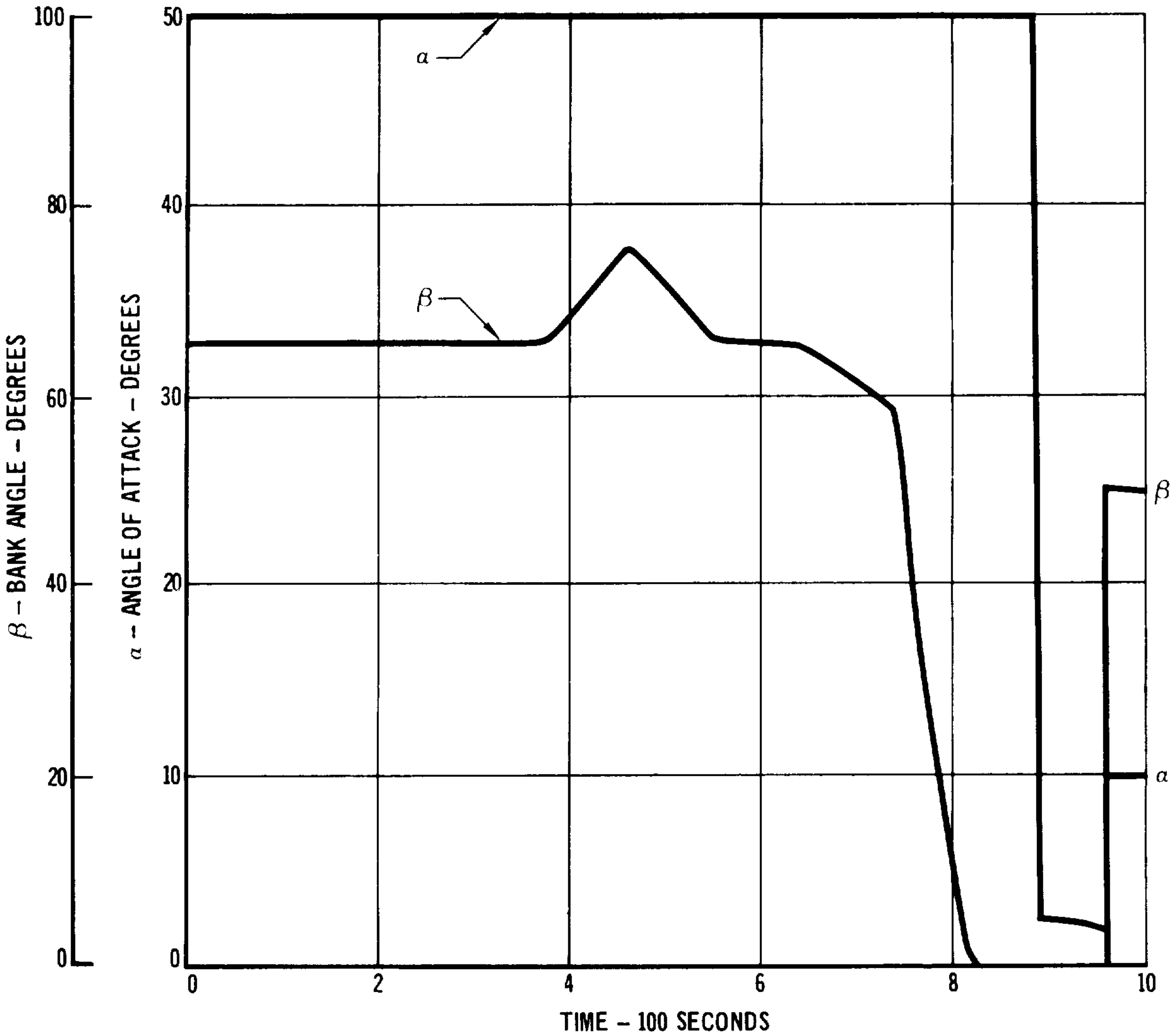


Figure 3–71

SECOND STAGE REENTRY TRAJECTORY COMPARISON
Altitude – Velocity Profile

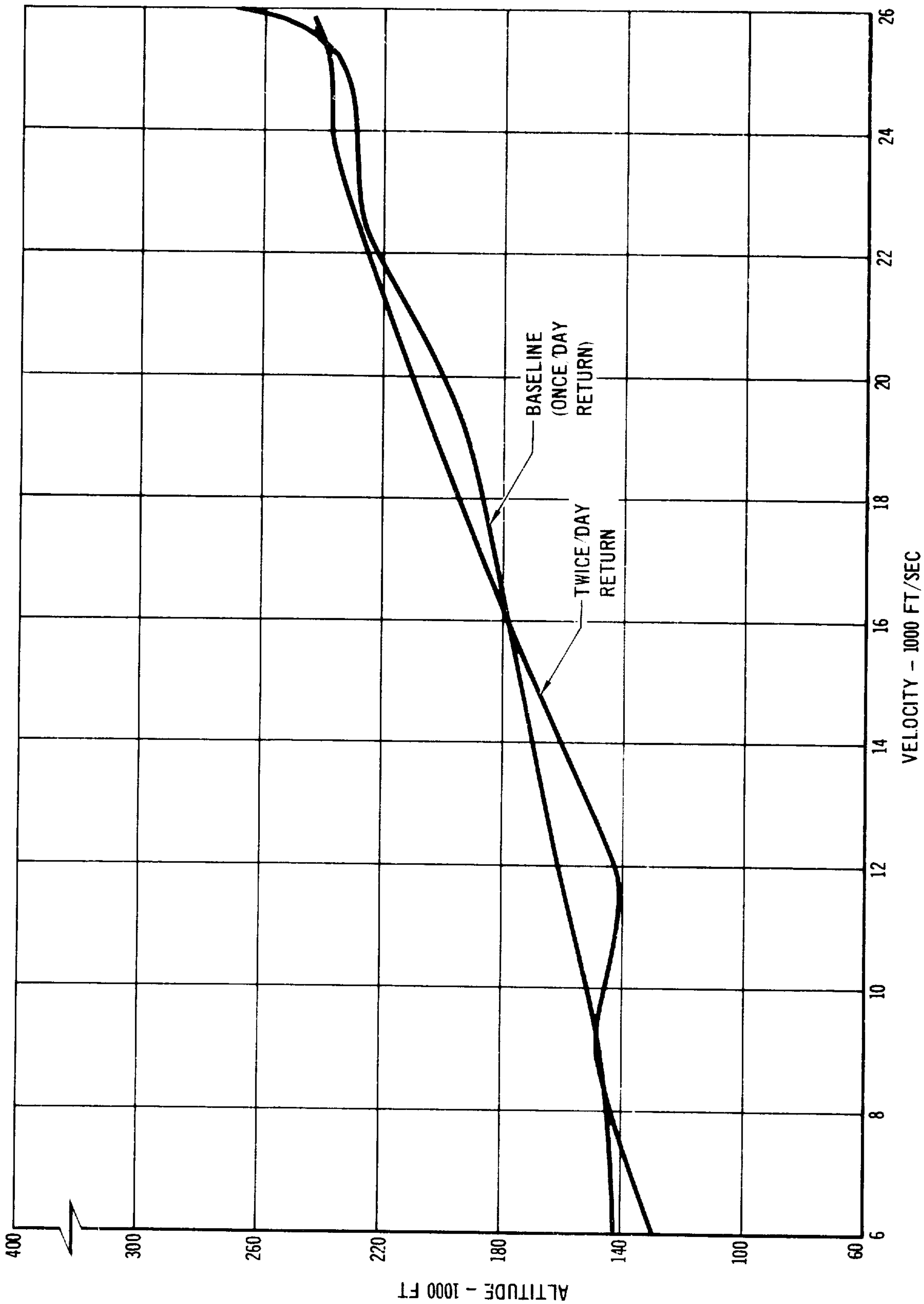
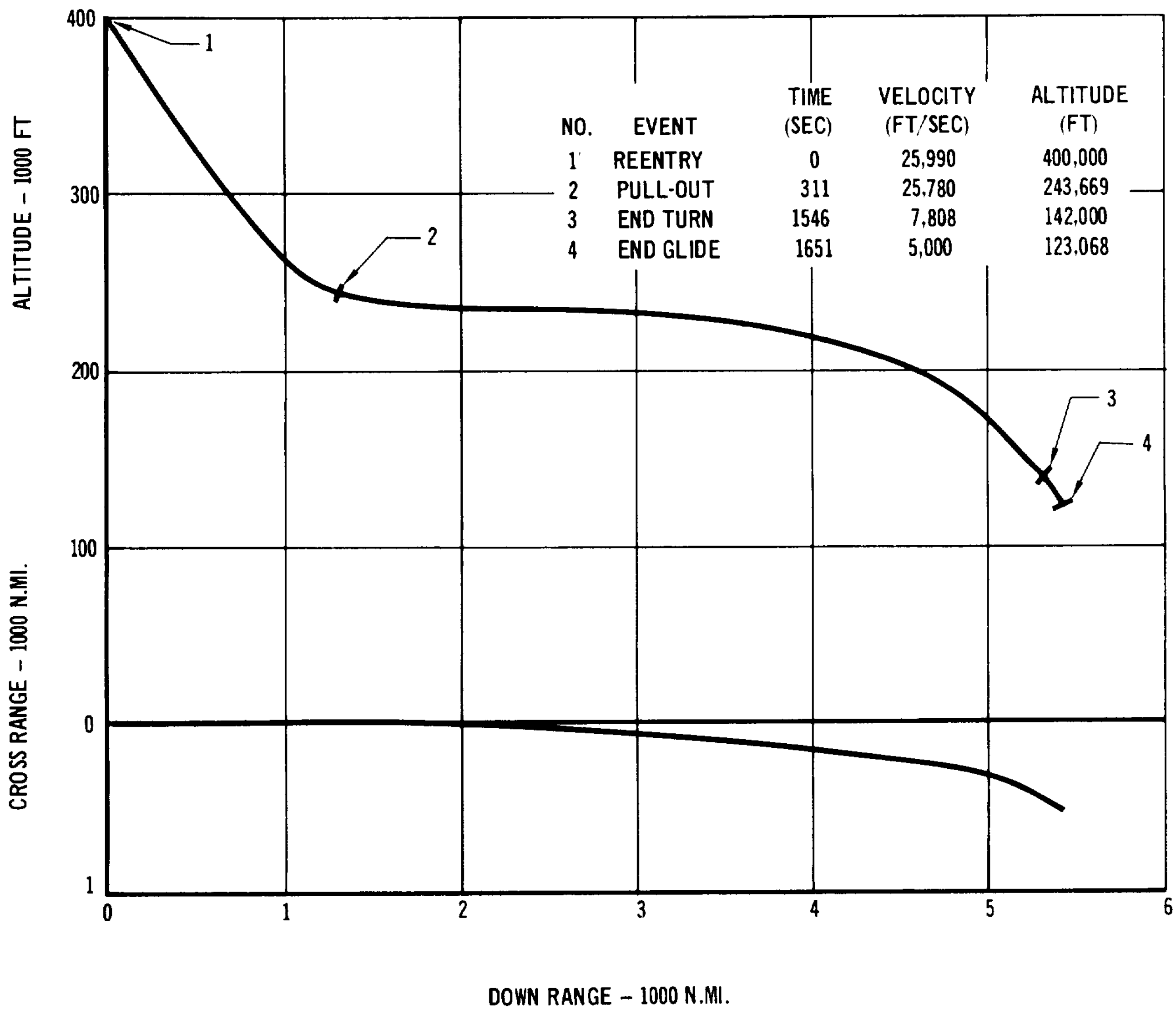


Figure 3-72

TWICE/DAY RETURN REENTRY



NO.	EVENT	TIME (SEC)	VELOCITY (FT/SEC)	ALTITUDE (FT)
1	REENTRY	0	25,990	400,000
2	PULL-OUT	311	25,780	243,669
3	END TURN	1546	7,808	142,000
4	END GLIDE	1651	5,000	123,068

Figure 3–73

TWICE/DAY RETURN REENTRY TRAJECTORY PARAMETERS
Part I

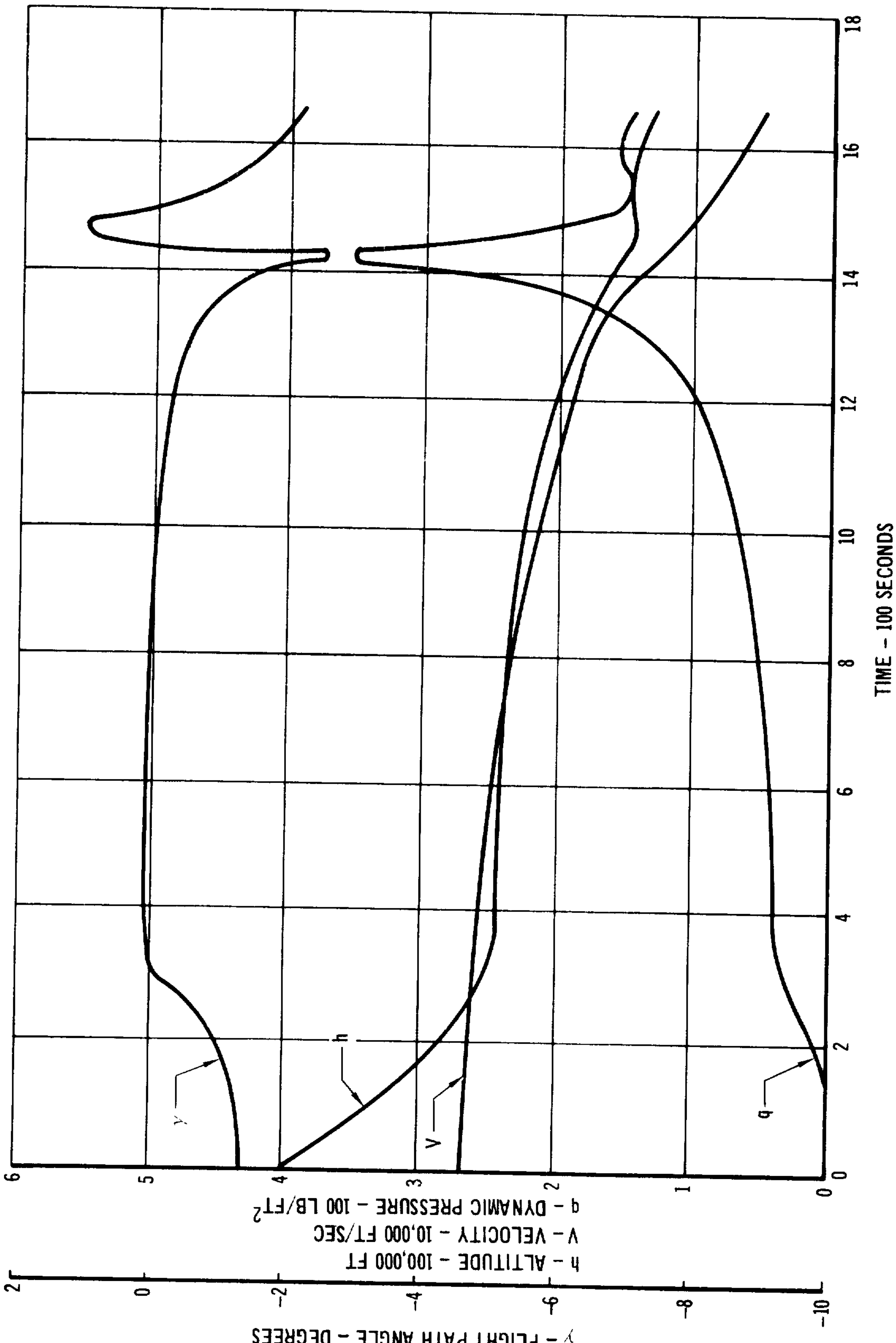


Figure 3-74

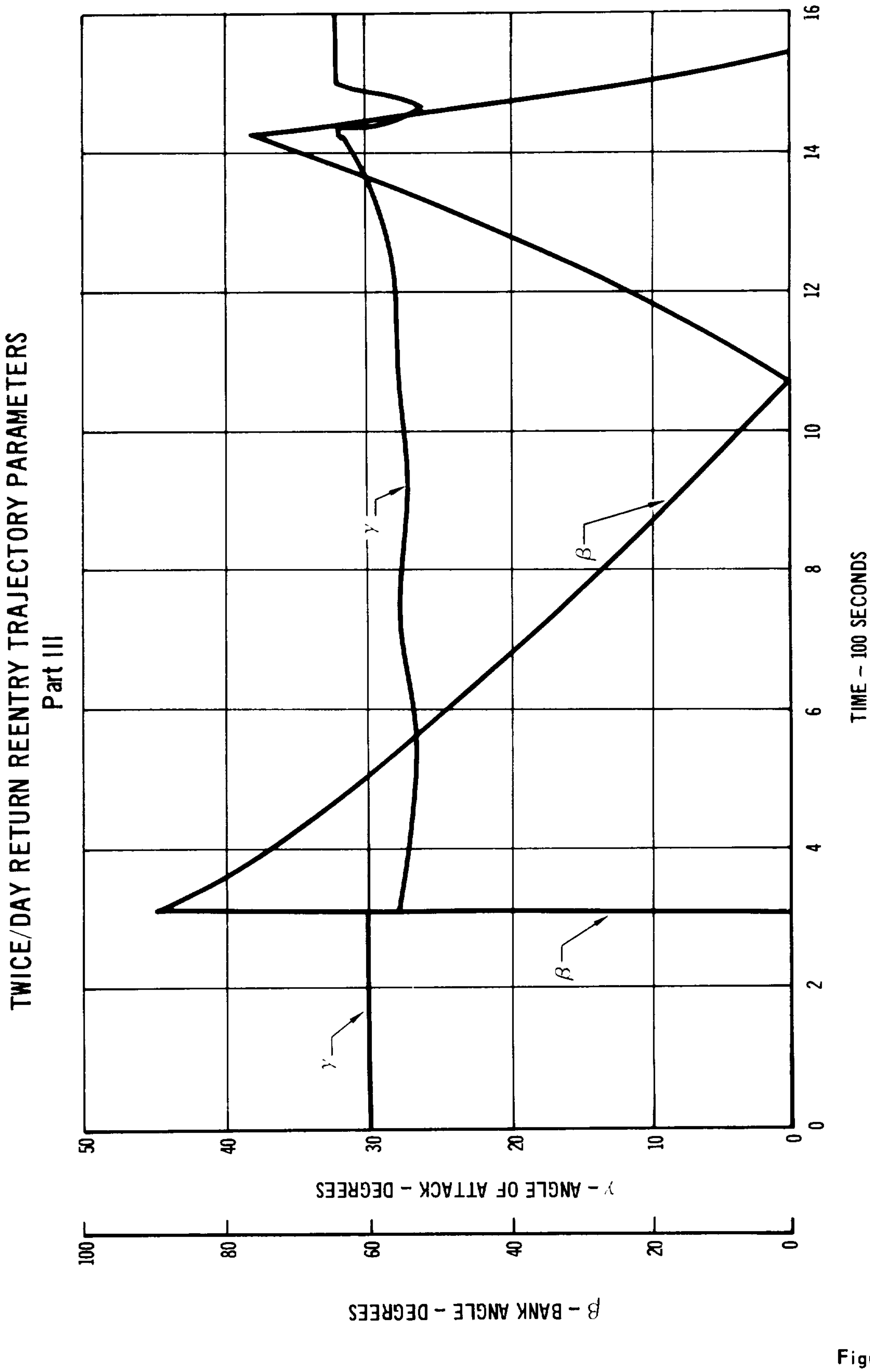

TWICE/DAY RETURN REENTRY TRAJECTORY PARAMETERS
Part III
γ
β
β
γ
TIME – 100 SECONDS
0
2
4
6
8
10
12
14
16
γ – ANGLE OF ATTACK – DEGREES
0
10
20
30
40
50
β – BANK ANGLE – DEGREES
0
20
40
60
80
100

Figure 3-75

TWICE/DAY RETURN REENTRY TRAJECTORY PARAMETERS
Part II

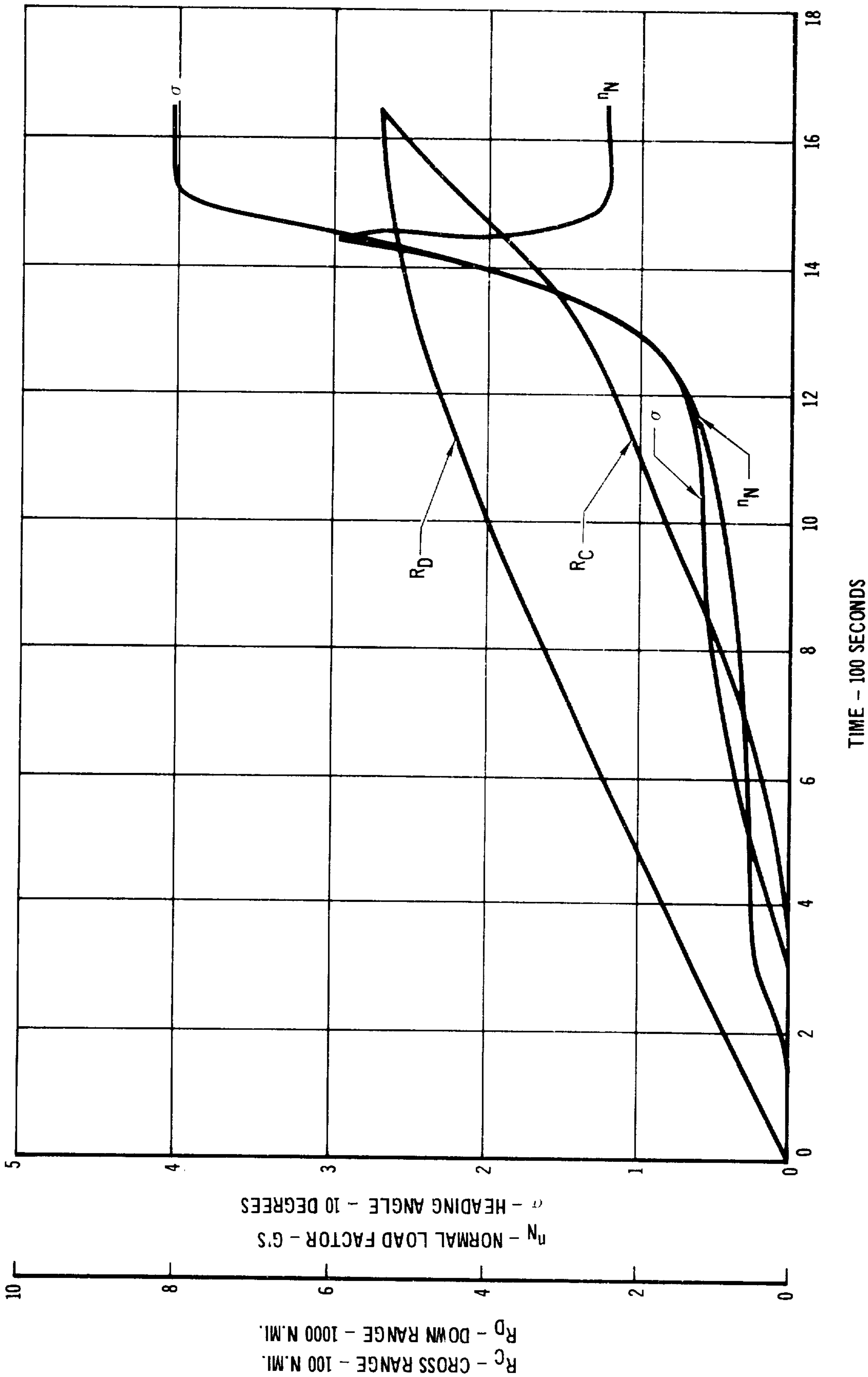


Figure 3-76

c) Orbiter Entry Heating Analysis - This section presents the Orbiter entry heating analysis, the methods used to compute entry heating, and the effects of turbulent heating on the Orbiter. Thermal protection requirements for Orbiter entry are presented in Volume I, Section 4.2.

- Aerodynamic Laminar Heating Methods-Orbiter - Local laminar heating rates for the Orbiter were obtained utilizing the NASA-LRC heat transfer test data of Reference (8). This heat transfer test data is presented in the ratios of local to stagnation point heat transfer coefficients ratios. It was assumed that $\left(\dot{q}_{SURFACE}/\dot{q}_{STAG}\right) \cong \left(h_{SURFACE}/h_{STAG}\right)$. Local heating rates on the Orbiter during flight were obtained by multiplying $\left(\frac{h_{SURF}}{h_{STAG}}\right)_{TEST}$ by the Fay and Riddell stagnation point heating rate on a hemisphere having a radius equal to the Orbiter nose radius (62 inches). Local radiation equilibrium temperatures were computed from $T_{SURF} = \left(\frac{\dot{q}_{SURF}}{\sigma\varepsilon}\right)^{1/4}$. A surface emittance of 0.85 was used to compute the radiation equilibrium temperatures. All temperatures presented are computed values and do not include an uncertainty factor.

The methods used to predict turbulent heating on the Orbiter are presented in the following section.

- Orbiter Entry Heating - Reentry heat pulses for the Orbiter are shown in Figure 3-77. Heat pulses are shown for the nominal once/day, minimum time (2600°F), twice/day and NASA-LRC $(C_L)_{MAX}$ reentries.

The twice/day reentry incurs the largest stagnation point total heat load (46,200 BTU/ft^2) and the minimum time (2600°F) reentry incurs the smallest (13,200 BTU/ft^2). The minimum time (2600°F) reentry has the highest stagnation point heating rate (59 BTU/ft^2 sec) and the NASA-LRC $(C_L)_{MAX}$ reentry the lowest (29 BTU/ft^2 sec).

Reentry temperatures on the Orbiter lower surface centerline at 25% of vehicle length are shown in Figure 3-78 for the four reentries. Peak temperatures during reentry are 2325°F for the minimum time (2600°F), 2100°F for the nominal once/day, 1725°F for the twice/day, and 1810°F for the NASA-LRC $(C_L)_{MAX}$.

ORBITER REENTRY HEAT PROFILES

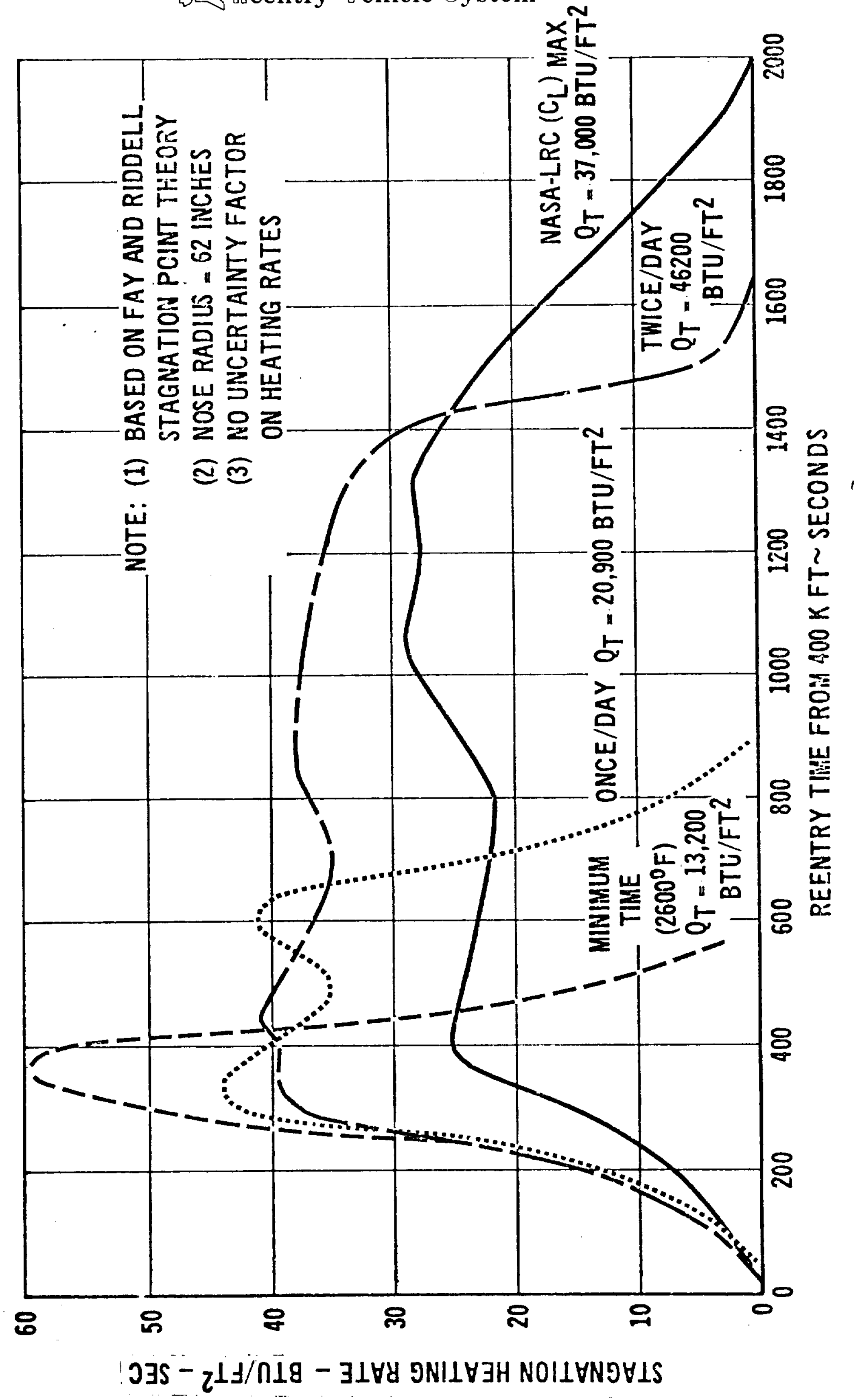


Figure 3-77

ORBITER REENTRY TEMPERATURES
Lower Surface Centerline
25% of Vehicle Length

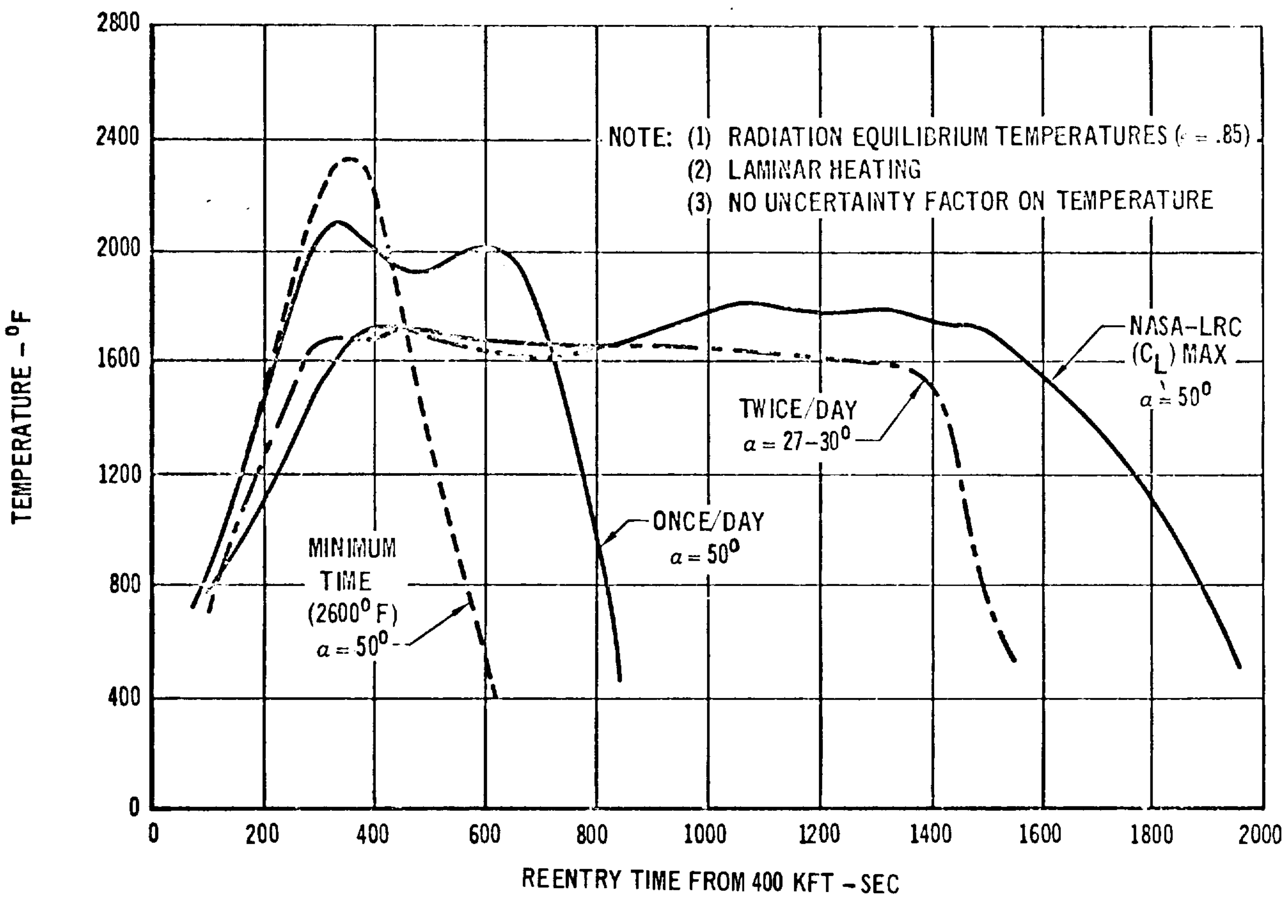


Figure 3-78

Maximum laminar radiation equilibrium Orbiter surface temperatures during reentry are shown in Figures 3-79 through 3-82 for the four reentries at four body stations.

Figure 3-79 shows that maximum surface temperatures for the nominal once/day reentry range from 680°F on the upper surface to 2200°F on the lower surface. Maximum surface temperatures for the minimum time (2600°F) reentry, Figure 3-80, range from 780°F on the upper surface to 2600°F on the lower surface while, maximum surface temperatures for the twice/day reentry, Figure 3-82, range from 700°F on the upper surface to 2200°F on the lower surface. Maximum surface temperatures for the NASA-LRC $(C_L)_{MAX}$ reentry, Figure 3-82, range from 550°F on the upper surface to 2110°F on the lower surface.

An Orbiter reentry heating comparison summary for the four reentries is presented in Figure 3-83. The figure shows the maximum stagnation point heating rate on a hemisphere having a radius equal to the orbiter nose radius (62 inches), the total stagnation point heat load, heating time and the range of the maximum surface temperatures. Maximum surface temperatures are approximately the same, except for the minimum time (2600°F) reentry. When time becomes an important factor, reentry can be accomplished with the associated high temperature penalty.

Turbulent Heating Effects on Orbiter - The use of a Reynolds number based on a local boundary-layer parameter, such as displacement or momentum thickness, is a method frequently used to estimate the onset of transition to turbulent flow. Such a parameter tends to correlate data very well. However, these correlations also tend to diverge (scatter of data increases) with decreasing local Mach number, and their use for a transition criterion is usually restricted to local Mach numbers greater than 3 or 4. Consequently, justification for the utilization of this type correlation for the Orbiter is questionable since the maximum local Mach number on the aft lower surface is less than 3 for $(L/D)_{MAX}$ flight conditions and less than 2 for $(C_L)_{MAX}$ flight conditions.

Utilization of a local boundary layer thickness parameter is also questionable because of the complex shape of the orbiter and the fact that the correlations are based on zero pressure-gradient surfaces (flat plates and cones). The lower surface of the orbiter is essentially a convex blunt delta wing. Theoretically, a transition parameter should be applicable regardless of pressure gradient effects.

ORBITER MAXIMUM TEMPERATURES (NOMINAL ONCE/DAY REENTRY)

NOTE:
(1) RADIATION EQUILIBRIUM TEMPERATURES (ε .85)
(2) LAMINAR HEATING
(3) NO UNCERTAINTY FACTOR
(4) α - 50^{0}

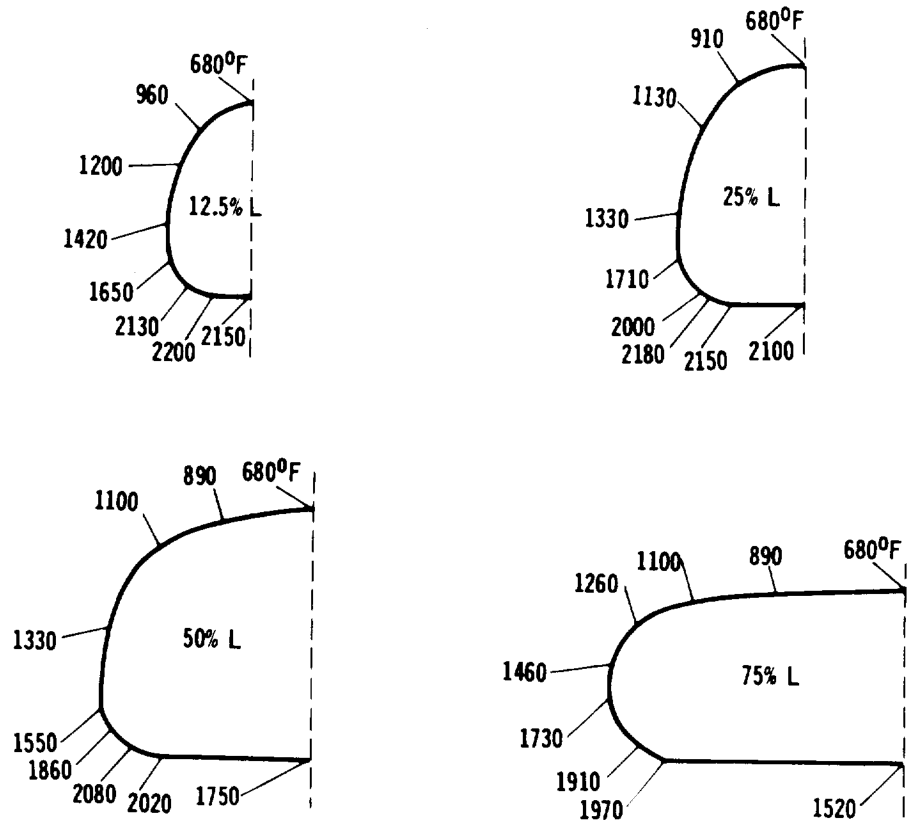
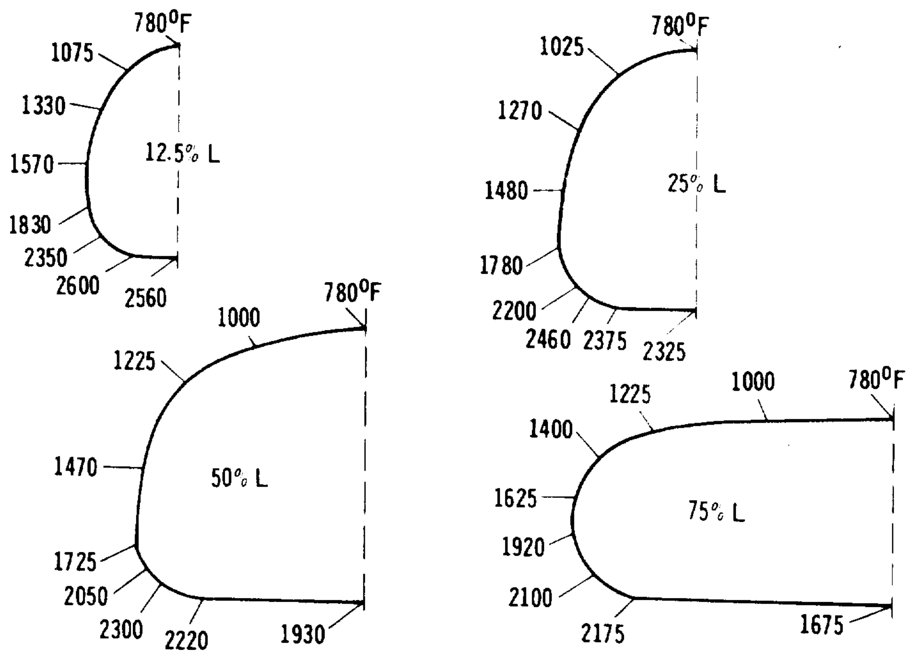


Figure 3–79

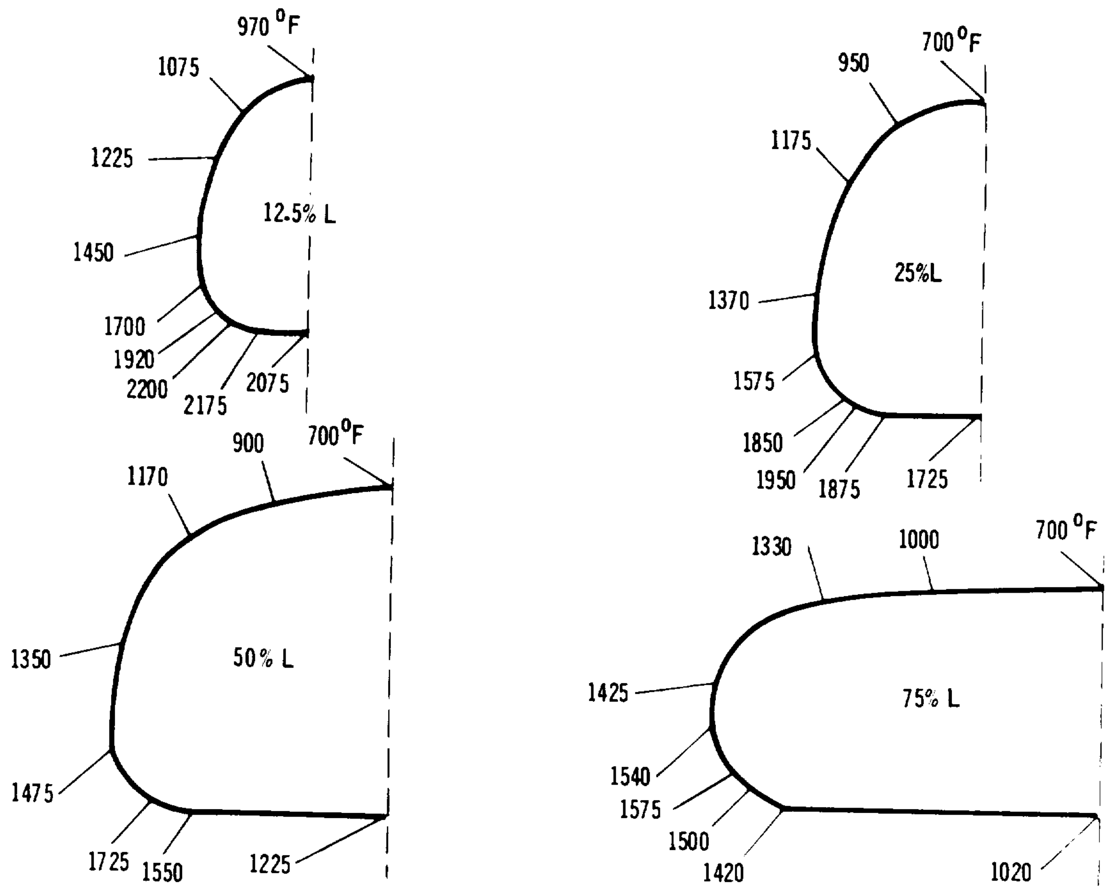
ORBITER
MAXIMUM TEMPERATURES
MINIMUM TIME (2600°F) REENTRY



NOTE:
(1) RADIATION EQUILIBRIUM TEMPERATURES(ε 0.85)
(2) LAMINAR HEATING
(3) NO UNCERTAINTY FACTOR
(4)α 50°

Figure 3-80

MAXIMUM TEMPERATURES TWICE/DAY REENTRY

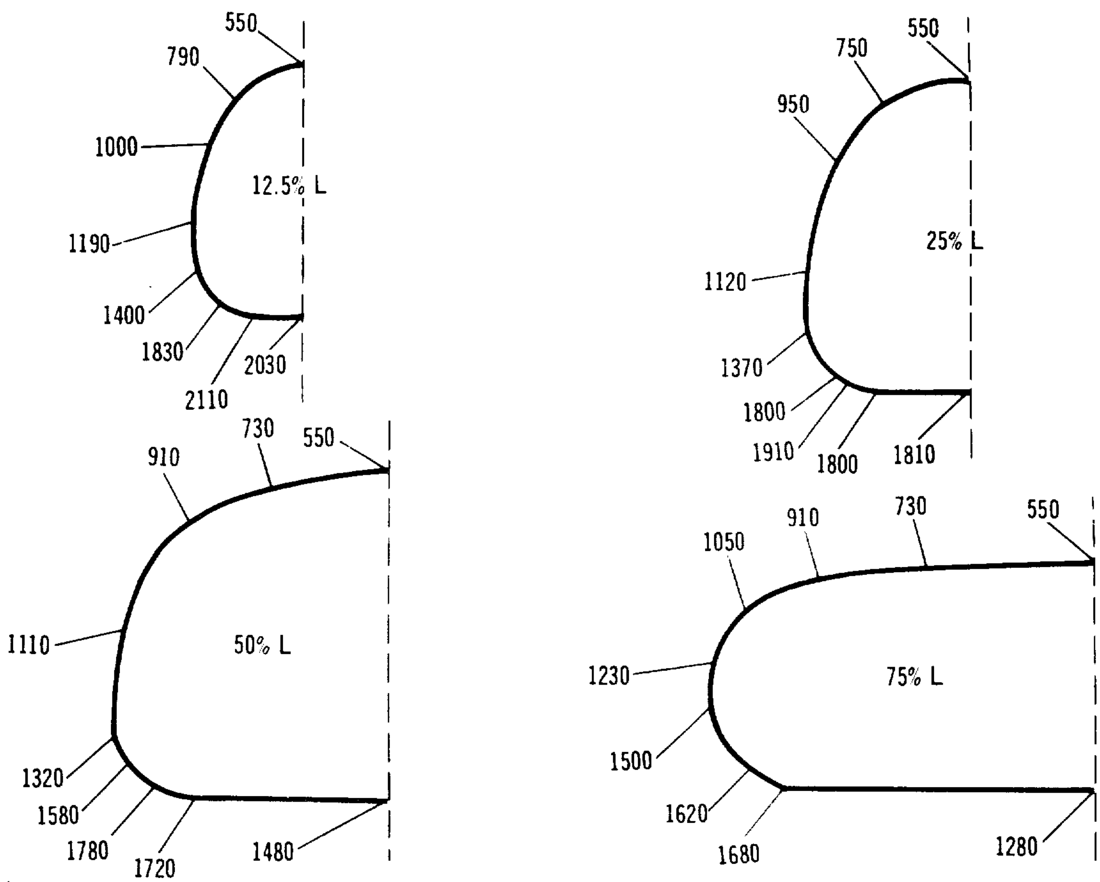


NOTE:
(1) RADIATION EQUILIBRIUM TEMPERATURES ($\varepsilon = 0.85$)
(2) LAMINAR HEATING
(3) NO UNCERTAINTY FACTOR
(4) α ~ $27-30^0$

Figure 3-81

MAXIMUM ORBITER TEMPERATURES
$(C_L)_{Max}$ Reentry

NOTE:
(1) RADIATION EQUILIBRIUM TEMPERATURES
($\epsilon = 0.85$)
(2) LAMINAR HEATING
(3) NO UNCERTAINTY FACTOR



ILRVS-351

Figure 3-82

ORBITER REENTRY HEATING COMPARISON

REENTRY	$(\dot{q}_0)_{MAX}$ (BTU/FT^2 SEC)	(Q_T) STAG (BTU/FT^2)	HEATING TIME (SECONDS)	MAXIMUM TEMPERATURE RANGE (^oF)
NASA MAX C_L – $\alpha = 50^o$	29	37,700	1,930	550 – 2110
TWICE/DAY – $\alpha \approx 27 - 30^o$	41	46,200	1,550	700 – 2200
ONCE/DAY – $\alpha = 50^o$	44	20,900	830	680 – 2200
MINIMUM TIME (2600^{o}F) – $\alpha = 50^o$	59	13,200	540	780 – 2600

NOTE: (1) $(\dot{q}_0)_{MAX}$ BASED ON RADIUS = 62 INCHES

(2) (Q_T) STAG AND HEATING TIME ARE FOR $q_0 > 1.0$ BTU/FT^2 SEC

(3) TEMPERATURES ARE PREDICTED LAMINAR RADIATION EQUILIBRIUM VALUES ($\varepsilon = .85$)

Figure 3-83

However, computation of the boundary-layer thickness parameter should also include this effect. The influence of outflow or spanwise pressure gradient should also be included for valid application of this transition criterion to the orbiter lower surface.

Computations of the nature described above are beyond the scope of the present study and transition was based on a local Reynolds number. The local Reynolds number on the lower surface centerline was determined by integrating the unit Reynolds number along the wetted distance to include pressure gradient effects on local flow properties as suggested in Ref. (9). Fully developed turbulent flow was assumed to occur at an integral local Reynolds number twice the value at the onset of transition. Turbulent heating rates were computed by the reference-enthalpy method using the integrated local Reynolds number. For angles of attack of interest in this study, this technique resulted in good agreement with the turbulent heating data of Ref. (9).

Selection of a criterion for transition on the Orbiter is somewhat difficult. From the Orbiter data in Ref. (9), only two transition points can be clearly defined. The two points correspond with integrated Reynolds numbers of approximately 0.5 and 0.7×10^6. Since these values were probably influenced by tunnel conditions and disturbances, onset of transition for the Orbiter in the present study was assumed to occur at an integrated Reynolds number of 10^6. However, transition onset at a value of 0.5×10^6 was also considered as a possibility for comparative purposes.

Variation of the integrated local Reynolds number with body station for the entry trajectories considered in this study are illustrated in Figures 3-84 and 3-85. The effects of turbulent heating on temperatures on the lower surface centerline at 50% vehicle length are shown in Figures 3-86 to 3-89 for the four reentries. A summary of the effects of turbulent heating for these reentries is presented in Figure 3-90.

Figure 3-90 shows that for fully developed turbulent flow at $Re_L = 2 \times 10^6$ maximum turbulent temperatures are less than maximum laminar temperatures, except for the twice/day entry. For the twice/day entry the maximum turbulent temperature is 215 degrees higher than the maximum laminar temperature. For fully developed turbulent flow at $Re_L = 10^6$ maximum turbulent temperatures are higher than maximum laminar temperatures for all entries except the NASA-LRC $(C_L)_{MAX}$. The increases in maximum temperatures for this criterion are 300 degrees for the nominal once/day entry, 470 degrees for the minimum time (2600°F) and 545 degrees for the twice/day. Fully developed turbulent flow at $Re_L = 10^6$ results in a shingle material change only for the twice/day entry where the increase in maximum temperature is from 1225°F (Rene'41) to 1770°F (TD-NiCr), since the maximum temperature limit for Rene'41 is 1600°F.

Figure 3-90 also shows that the effect of turbulent heating on total heat (Q_T) ranges from an increase of 4% for the NASA-LRC $(C_L)_{MAX}$ reentry to 43% for the twice/day reentry, with fully developed turbulent flow at $Re_L = 2 \times 10^6$ and 1×10^6, respectively. The thermal protection requirements shown in Volume I, Section 4.2 are increased by only a small amount for fully developed turbulent flow at $Re_L = 2 \times 10^6$ since thermal protection requirements are more strongly influenced by heating time than total heat. However, for fully developed turbulent flow at $Re_L = 10^6$ a significant increase in thermal protection requirements result, except for the NASA-LRC $(C_L)_{MAX}$ reentry which has an increase in total heat of only 8%.

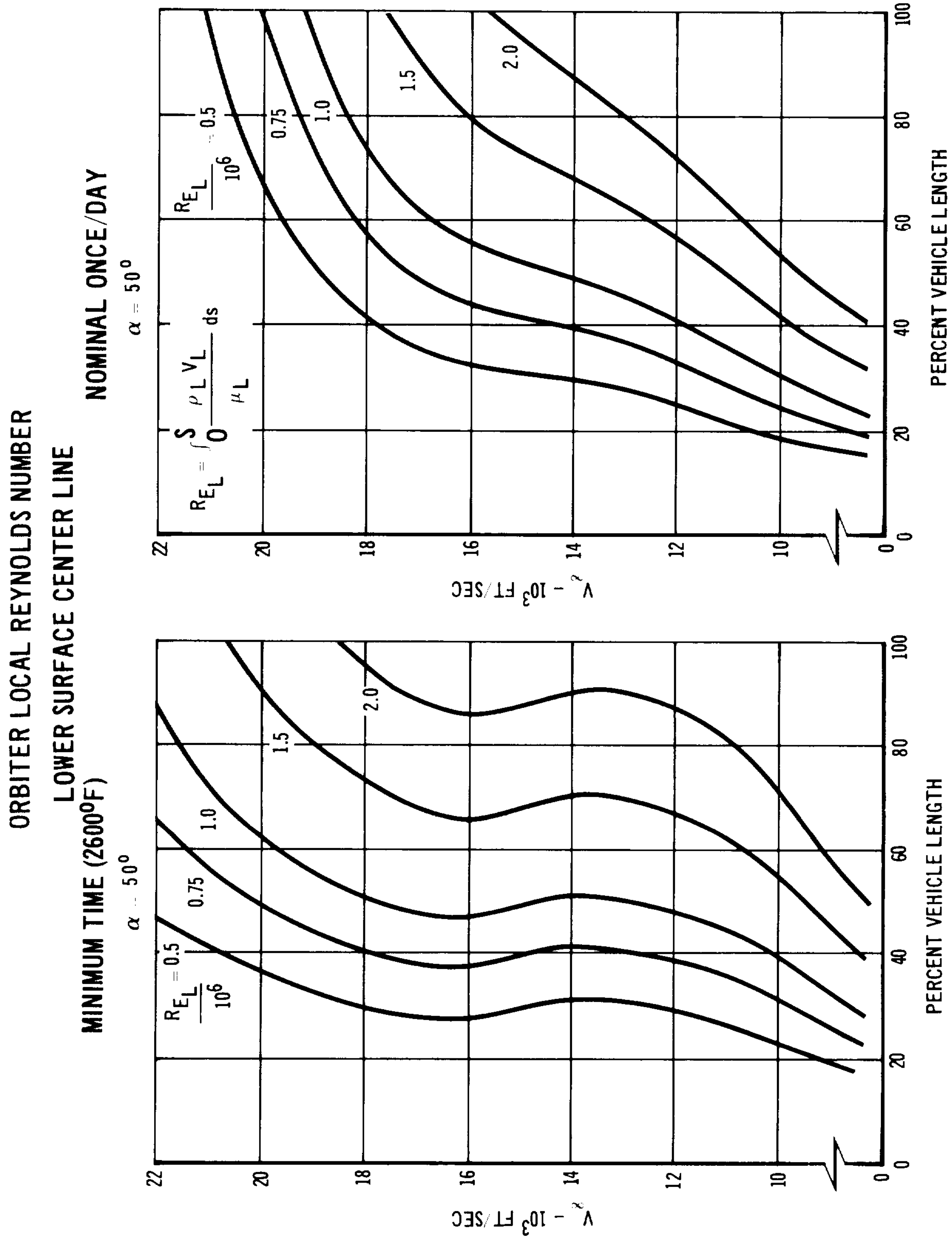

ORBITER LOCAL REYNOLDS NUMBER
LOWER SURFACE CENTER LINE
MINIMUM TIME (2600°F)
α = 50°
NOMINAL ONCE/DAY
α = 50°
$R_{E_L} = \int_0^S \frac{\rho_L V_L}{\mu_L} ds$
$\frac{R_{E_L}}{10^6} = 0.5$
0.75
1.0
1.5
2.0
PERCENT VEHICLE LENGTH
V_∞ – 10^3 FT/SEC
0
20
40
60
80
100
10
12
14
16
18
20
22

Figure 3-84

ORBITER LOCAL REYNOLDS NUMBER
(LOWER SURFACE CENTER LINE)

NASA-LRC $C_{L_{MAX}}$
$\alpha = 50^0$

TWICE/DAY
$\alpha = 27-30^0$

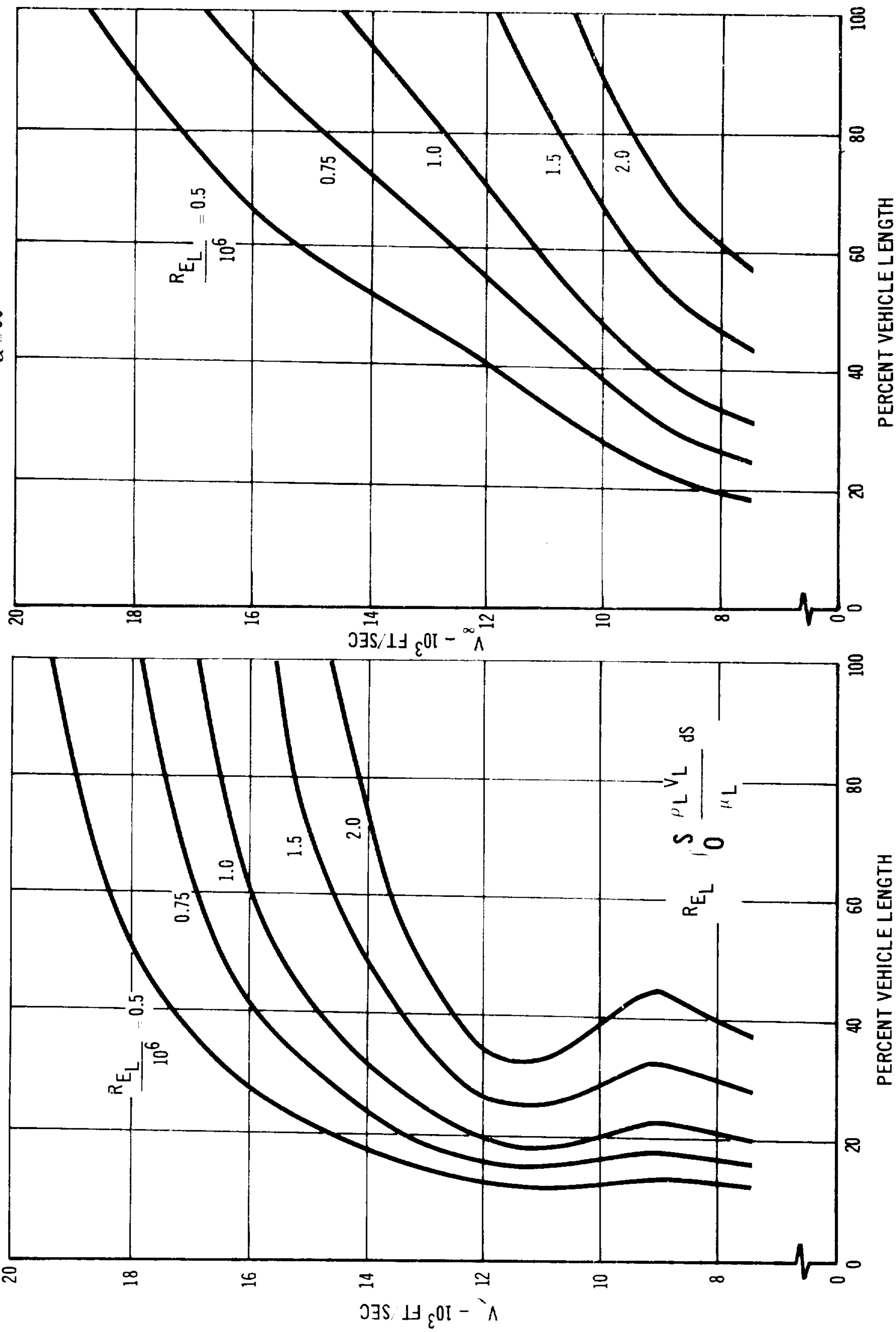


Figure 3-85

TRANSITION EFFECTS ON ORBITER TEMPERATURES

Once/Day Reentry

Lower Surface Centerline at X/L = 0.50

NOTE:
(1) RADIATION EQUILIBRIUM TEMPERATURES (ε .85)
(2) NO UNCERTAINTY FACTOR

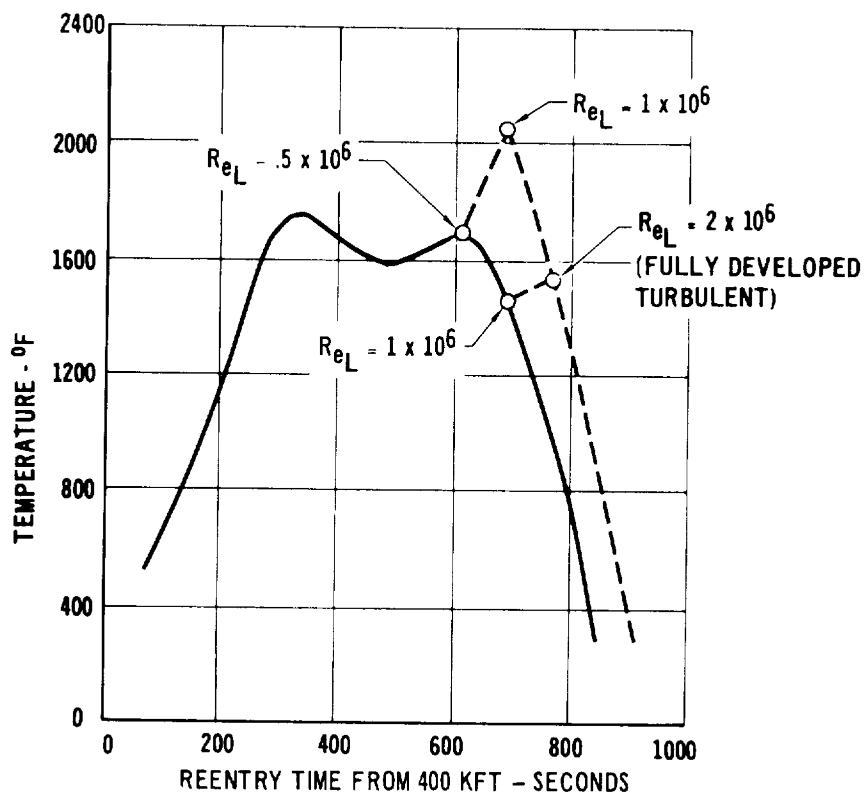


Figure 3–86

TRANSITION EFFECTS ON ORBITER TEMPERATURES
Minimum Time (2600 °F) Reentry
Lower Surface Centerline at X/L = .50

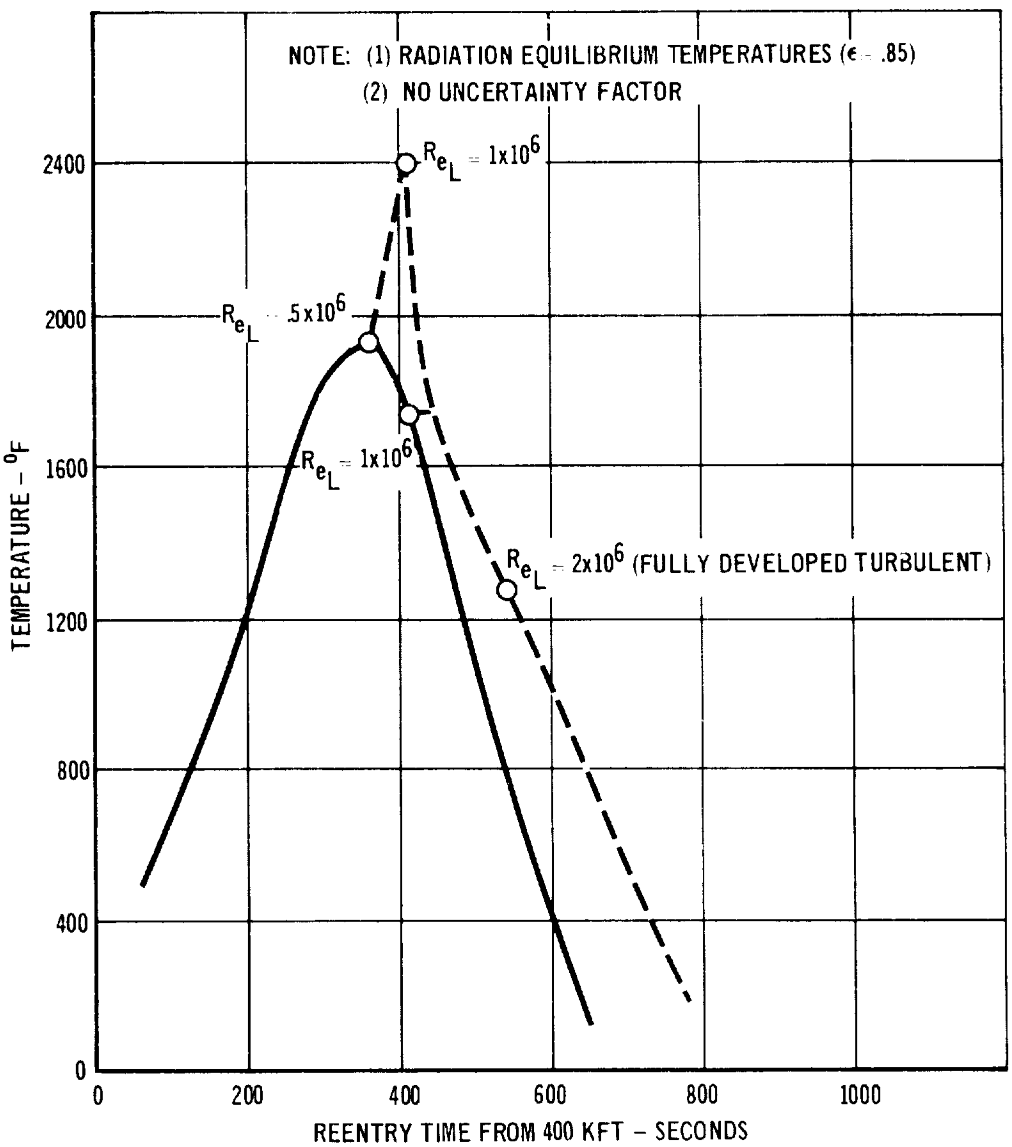


Figure 3-87

TRANSITION EFFECTS ON ORBITER TEMPERATURES
Twice/Day Reentry
Lower Surface Centerline, X/L = .50

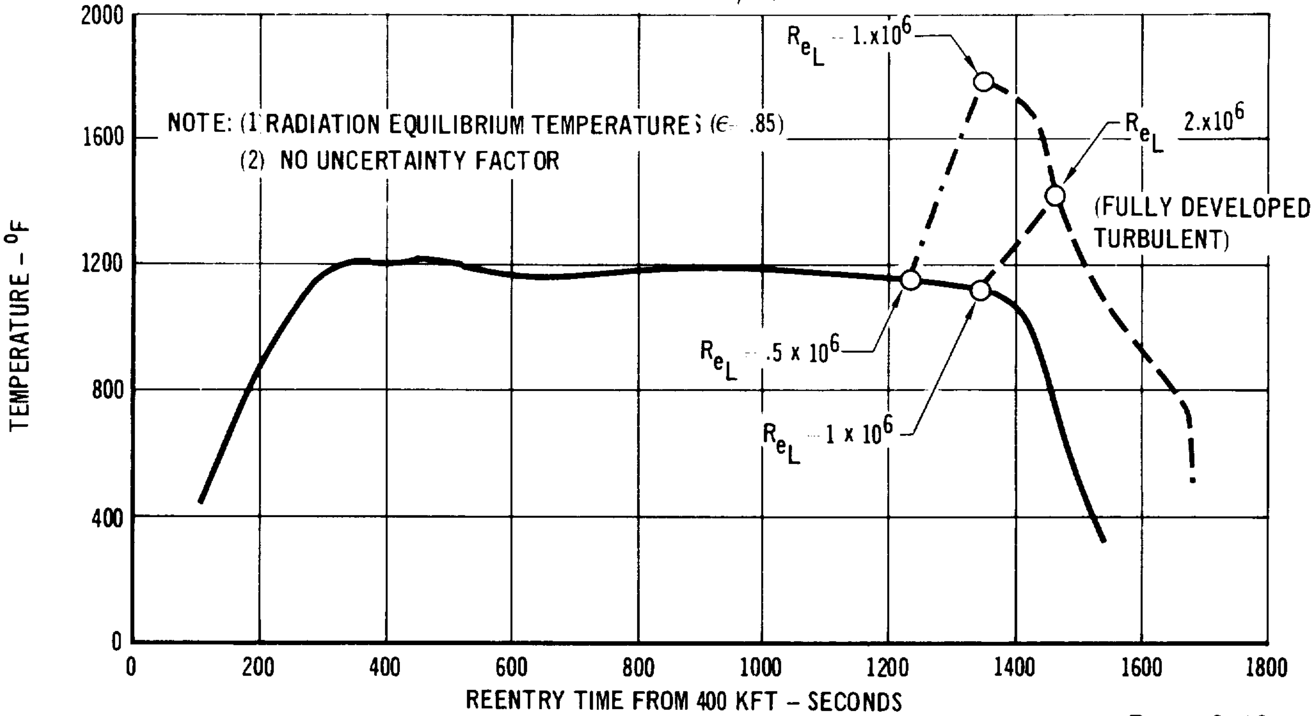


Figure 3-88

TRANSITION EFFECTS ON ORBITER TEMPERATURES
NASA-LRC $C_{L_{MAX}}$ Reentry
Lower Surface Centerline at X/L = .50

NOTE: (1) RADIATION EQUILIBRIUM TEMPERATURES (ϵ = .85)
(2) NO UNCERTAINTY FACTOR

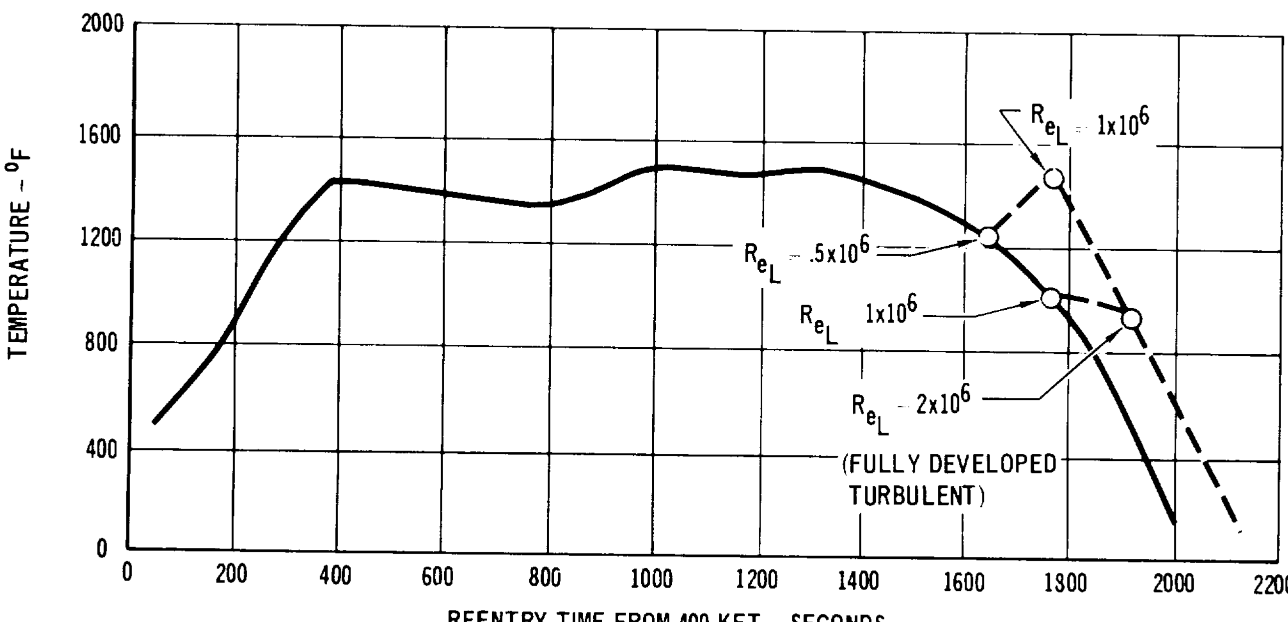


Figure 3-89

EFFECT OF TURBULENT HEATING ON ORBITER
Lower Surface Centerline, X/L = .50

REENTRY	T_{MAX}, LAMINAR	T_{MAX}, TURB (1)	T_{MAX}, TURB (2)	Q_T, LAM (BTU/FT^2)	INCREASE IN Q_T DUE TO TURB HEATING	
					$R_{E_L} = 2 \times 10^6$ (1)	$R_{E_L} = 10^6$ (2)
ONCE/DAY	1750 °F	1540 °F	2050 °F	4280	7%	27%
MIN TIME (2600 °F)	1930	1270	2400	2710	16	40
TWICE/DAY	1225	1440	1770	3700	17	43
NASA-LRC $C_{L_{MAX}}$	1480	940	1470	7725	4	8

NOTE: (1) TRANSITION ONSET AT $R_{E_L} = 10^6$, FULLY DEVELOPED TURBULENT AT $R_{E_L} = 2 \times 10^6$

(2) TRANSITION ONSET AT $R_{E_L} = .5 \times 10^6$, FULLY DEVELOPED TURBULENT AT $R_{E_L} = 10^6$

Figure 3–90

3.4 Approach and Landing - Following reentry, either from orbit (Orbiter) or staging (Carrier), each vehicle must be able to land horizontally. The type of approach utilized must be tailored to the entry trajectory, vehicle performance capability (power vs unpowered), landing conditions (VFR vs IFR) and mode of operation (manual or automatic). A special emphasis study, addressing itself to these questions was performed and is presented in this section. The details of the approach and landing maneuvers are presented for each vehicle. A powered approach with go-around capability was a groundrule for this study. Various go-around systems options were investigated together with the payload penalties associated with each option. The concept of an automatic landing system was also investigated and digital computer simulations for automatic approaches and landings are presented for both the Orbiter and Carrier.

3.4.1 Orbiter Approach - The Orbiter does not have an extended cruise capability but has adequate footprint to reach a high key position from an unpowered glide. Airbreathing landing assist engines are deployed and started upon reaching the high key position (30,000 ft. altitude and Mach 0.64). The high key point is selected to enable an idle power VFR descent. The same high key position is used for the IFR approach so that an unpowered, dead-stick approach may be made in the event that the landing assist engines cannot be deployed or started. The idle power approach is similar in technique to that employed by the X-15, HL-10 and other unpowered, low L/D vehicles in landings at the Flight Research Center (FRC) and is initiated at the high key position above the landing site followed by a spiral descent in which calibrated air speed and bank angle are kept constant. Following 360 degrees of turn, the vehicle is rolled out onto its final approach followed by a high energy flare and subsequent landing. Pilots of unpowered low L/D vehicles prefer this approach over the straight-in approach because they can maintain continual visual contact with their intended touchdown point and by utilizing speed brake control and angle-of-attack modulation can manage the potential and kinetic energy to correct for adverse winds and insure that the runway is reached. Touchdown accuracies of ± 700 feet have been recorded at FRC in recent HL-10 landings, followed by a roll-out distance of under 7000 feet utilizing moderate wheel braking. While the orbiter has the demonstrated capability to make a completely unpowered approach and landing, the final approach may be made on a shallower glide path (e.g., eight degrees) if desired, utilizing power assist.

A typical 360 degree high key overhead approach, initiated at 30,000 feet and Mach 0.64, is shown in Figures 3-91 and 3-92. During the descent the velocity and bank angle are kept constant at 240 KCAS and 45 degrees respectively until roll-out for either the downwind leg of the IFR approach or the final approach of the VFR approach. The VFR approach shown is unpowered, representing the type of approach that could be made in the event that landing assist propulsion is not available. The IFR approach requires power assist beginning at the turn onto the base leg (t = 99 seconds) and continuing through the subsequent final descent which is made on a three degree glide slope at a calibrated airspeed of 200 knots. The downwind leg of the IFR approach is made at idle power with a glide angle of approximately 15 degrees until an altitude of 8000 feet is reached at which time a 180 degree turn is made onto the final approach. The final approach is made on a three degree glide slope initiated at 2000 feet over the outer marker which is 6.3 na mi from the end of the runway.

Although the approach shown requires 360 degrees of turn, this approach may be easily modified to a 270 degree approach to allow approaches from directions perpendicular to the runway. Similarly, the landing direction may be changed by 180 degrees by making the second and the subsequent turns to the right instead of the left. This maneuver will provide the capability to always land into the wind, regardless of initial approach direction. The high key point for the 270 degree approach is superimposed on Figure 3-91 and occurs at an altitude of approximately 23,000 feet over the intended touchdown point.

3.4.2 Landing Options - Four approach and/or go-around options have been considered in this study. They are: (1) VFR landing assist; (2) IFR powered approach; (3) 360 degree turn at 2000 feet altitude; (4) wave-off. These options may be utilized individually or in combination as indicated in Figure 3-93 which also presents the resulting incremental changes in payload capability. The simplest option, VFR landing assist, does not provide for go-around but merely provides intermittent glide slope control capability to reduce the possibility of a go-around. The second option, powered approach, provides adequate fuel to make an IFR approach. The third option provides a 360 degree turn capability at an altitude of 2000 feet and could be used to acquire a corrected approach pattern. The last option, wave-off, considers a climb-out from an altitude of 50 feet after wave-off with subsequent go-around and reacquisition of the outer marker. This

ORBETER APPROACH
(Plan View)

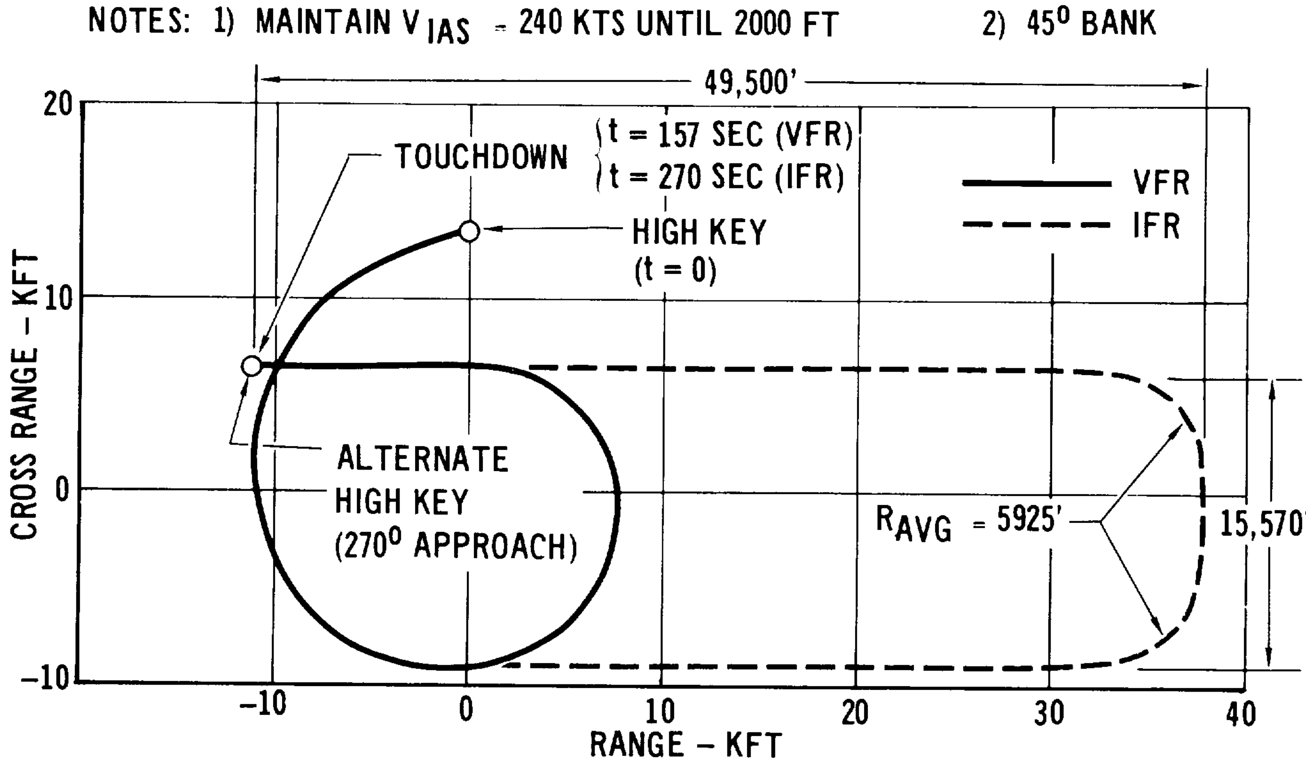


Figure 3-91

ORBITER APPROACH
(Elevation View)

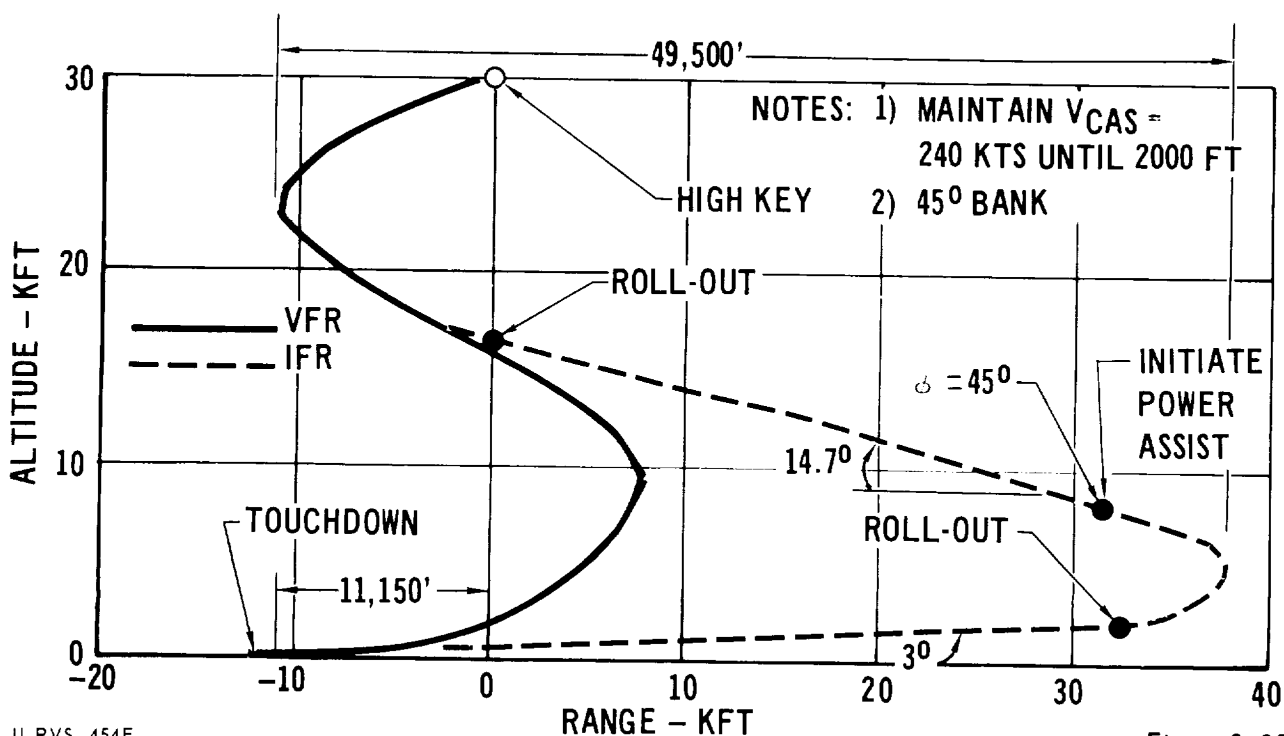


Figure 3-92

LANDING OPTIONS
Orbiter

OPERATIONAL OPTIONS				EFFECTS	
LANDING** ASSIST	POWERED** APPROACH	360^0 TURN AT 2000 FT	WAVE* OFF	THRUST/WT (REQ'D)	PAYLOAD INCREMENT (LB)
√				0.10	+ 16,800
	√			0.24	+ 7,500
	√	√		0.28	+ 4,000
	√		√	0.33	0 (BASELINE)
	√	√	√	0.33	− 3,500
√			√	0.33	+ 1,200

* CLIMB-OUT FROM 50 FT ALTITUDE AND RETURN TO OUTER MARKER

** WITH ENGINE OUT

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Figure 3–93

last option represents the selected baseline. Changes in payload capability (from the baseline) resulting from alternate options vary from minus 3500 lbs. to plus 16,800 lbs.

Go-around Pattern - The Orbiter go-around pattern geometry and corresponding sequence of events are shown in Figure 3-94. The go-around engines were sized to produce level flight (i.e., thrust = drag) at 2000 feet altitude on a standard day with one engine out (see Vol. I, Book 2, Section 3.4.2). The go-around pattern was computed assuming four engine operation, thus thrust = (4/3) (drag) and the excess power provides a rate of climb capability which is given by the expression

$$\frac{dh}{dt} = \frac{V}{(L/D)} \left[\frac{4}{3} - \frac{1}{\cos \phi} \right] \frac{ft}{sec},$$

where: V = velocity, ft/sec

L/D = lift-drag ratio

ϕ = bank angle, degrees.

Thus, the maximum sea level rate of climb corresponding to Mach = 0.34 and a lift-drag ratio of 4.3 is 29.1 feet per second or 1747 feet per minute. The rate of climb and altitude time histories during the go-around pattern are shown in Figure 3-95. Following a wave-off, which is shown occurring immediately over the end of a 10,000 foot runway, climb-out is made with gear up and at the maximum power setting until directly over the end of the runway. A 20 degree banked, co-ordinated, climbing turn to the left is then made until the 2000 foot pattern altitude is reached. The turn is continued at constant altitude until the down-wind leg is reached. Approximately 7 na mi beyond the end of the runway, a continuous 180 degree turn is made onto the final approach, at which time the landing gear is lowered and the vehicle decelerated so that it intercepts a three degree glide slope over the outer marker which is 6.3 na mi from the end of the runway. The final descent is made on this glide slope until touchdown.

The time histories of thrust and fuel expended during the Orbiter go-around are presented in Figure 3-96, showing that 7300 lbs. of fuel are required for this maneuver. An additional 2000 lbs of fuel are required for the IFR approach (see Figure 3-92). Thus, together with the go-around requirement, a total fuel weight of 9300 lbs. is required in order to make an IFR approach and go-around. These computations are based upon a vehicle weight of 199,160 lbs. and a specific fuel consumption of 1.15 lbs. of propellant per lb. of thrust per hour.

ORBITER GO-AROUND PATTERN SEQUENCE OF EVENTS

EVENT	TIME AT INITIATION OF EVENT REFERENCED TO WAVE OFF, SEC.
① CLIMB OUT AT $\dot{h} = 29.1$ FT/SEC; $\gamma = 4.44^0$	0
② 20^0 BANKED TURN; $\dot{h} = 22.6$ FT/SEC; $\gamma = 3.44^0$	27
③ LEVEL FLIGHT 20^0 BANKED TURN	80
④ CRUISE AT $h = 2{,}000$ FT – $V_{CAS} = 218$ KTS	127
⑤ LEVEL FLIGHT 20^0 BANKED TURN	243
⑥ CONSTANT ALTITUDE DECELERATION	344
⑦ DESCEND ON 3^0 GLIDE SLOPE; $\dot{h} = -17.7$ FT/SEC	359
⑧ TOUCHDOWN AT 164 KTS	472

ORBITER GO-AROUND PATTERN

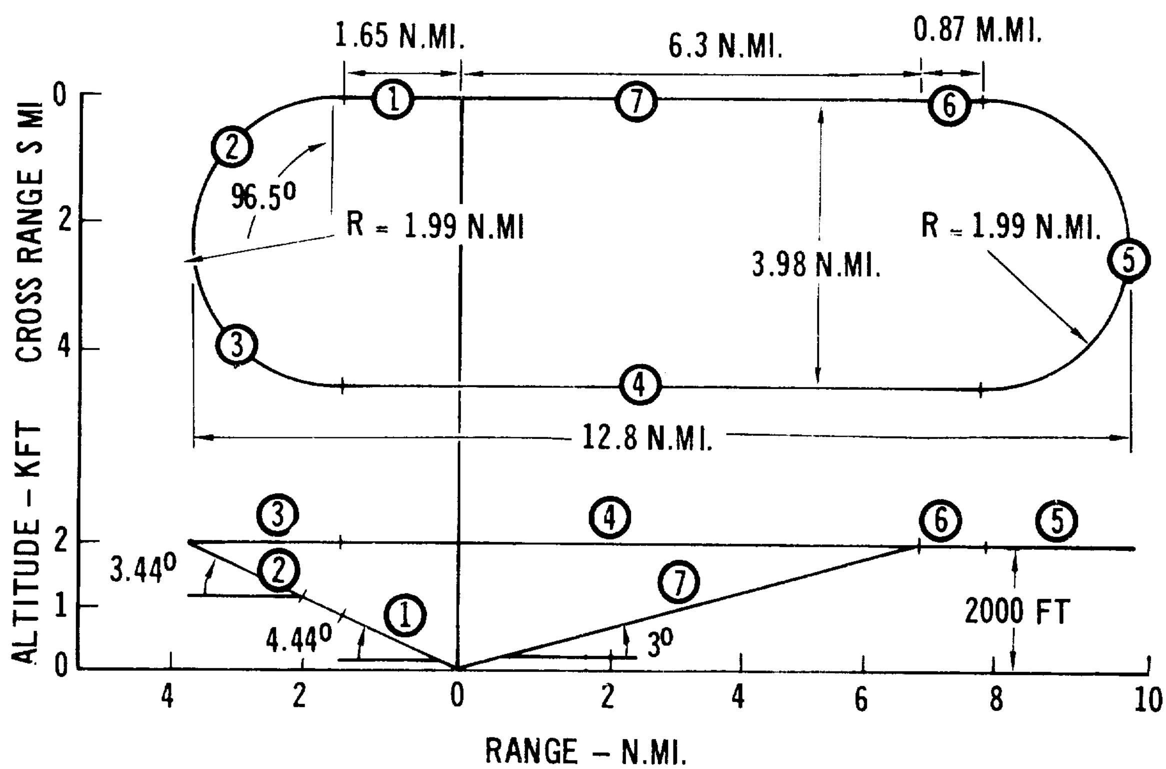

Figure 3-94

ORBITER GO-AROUND ALTITUDE AND RATE OF CLIMB
TIME HISTORY

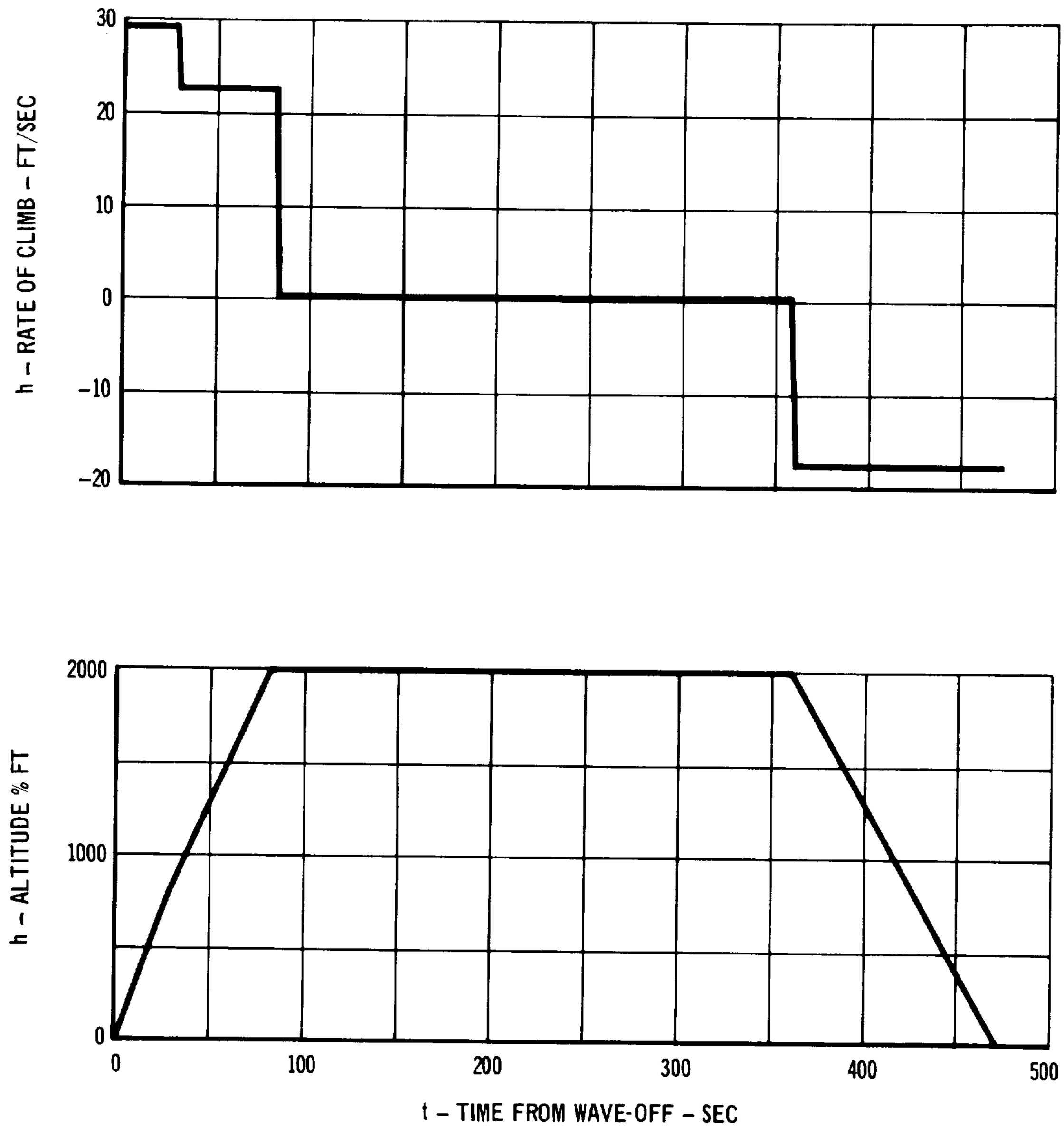

Figure 3-95

ORBITER GO-AROUND THRUST AND FUEL REQUIREMENTS

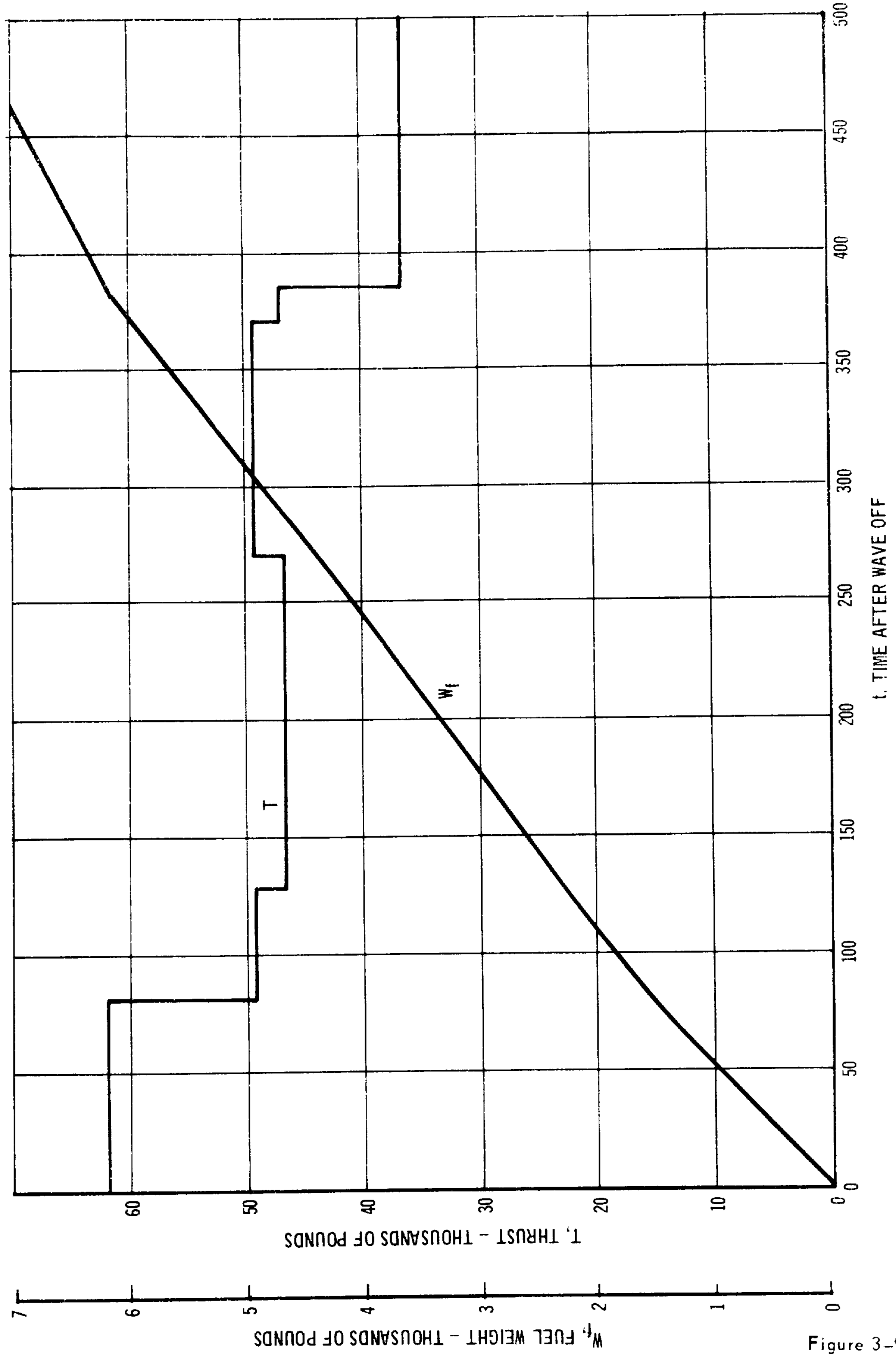

Figure 3-96

3.4.3 Orbiter Landing - The landing sequence consists of final approach and touchdown. These considerations are discussed in this order.

Final Approach - The final approach is made with a high excess velocity (e.g., 70 knots), executing a high altitude flare, followed by the float to touchdown. For a given flare load factor, the flare initiation altitude is directly proportional to the excess velocity which is the difference between the approach and touchdown velocities. Figure 3-97 shows a typical "high energy" approach and landing corresponding to an approach velocity of 240 knots and a maximum lift-drag ratio (with landing gear down) of 3.7. Following roll-out from the spiral descent (2000 foot altitude), the Orbiter is trimmed so that its flight path is aligned to intersect a point which is 0.6 na mi from the intended touchdown point. Upon reaching an altitude of 575 feet, the landing gearis lowered and a constant 1.5g normal load factor flare is executed until a flight path angle of -3 degrees is attained. The seubsequent descent (float) is made on a 3 degree glide slope which is matained until touchdown by angle of attack modulation.

Touchdown - The Orbiter landing speed variations with angle of attack are shown in Figure 3-98 for a wing loading of 47.7 lbs/ft^2 at sea level with and without ground effect. At a design touchdown angle of attack of 23 degrees the estimated ground effect increases the lift curve slope by eight percent, resulting in a touchdown speed of 168 knots. The above wing loading is based upon the entry weight (195,765 lbs.). Following an IFR go-around, the reduced wing loading (45.3 lbs/ft^2) will result in a touchdown speed of 164 knots.

3.4.4 Carrier Approach - Carrier cruiseback altitude is 10,000 feet at Mach 0.3. The carrier approach is initiated 17.4 na mi from the runway and a straight-in idle power descent is made along an eight degree glide slope, maintaining a constant calibrated airspeed of 175 kts. This path will intersect 2000 foot altitude over the outer marker, located 6.3 na mi from the runway. The final (IFR) approach is made on a 3 degree glide slope at a calibrated airspeed of 160 knots. The approach trajectory is shown on Figure 3-99.

A VFR approach is made in essentially the same manner with the exception that the final approach may be initiated somewhat closer to the runway and the approach descent made at a higher rate of sink.

Adequate fuel is provided to enable the Carrier to overfly the landing site and make an IFR final approach from the opposite direction.

ORBITER VFR (IDLE POWER) FINAL APPROACH

EVENT	TIME FROM TOUCHDOWN (SEC)
① ROLL-OUT ON FINAL APPROACH – V_{IAS} = 240 KTS	30
② INITIATE 1.5g FLARE; LOWER LANDING GEAR	15
③ INITIATE FLOAT PERIOD – V_{IAS} = 214 KTS	8
④ TOUCHDOWN – V_{IAS} = 168 KTS	0

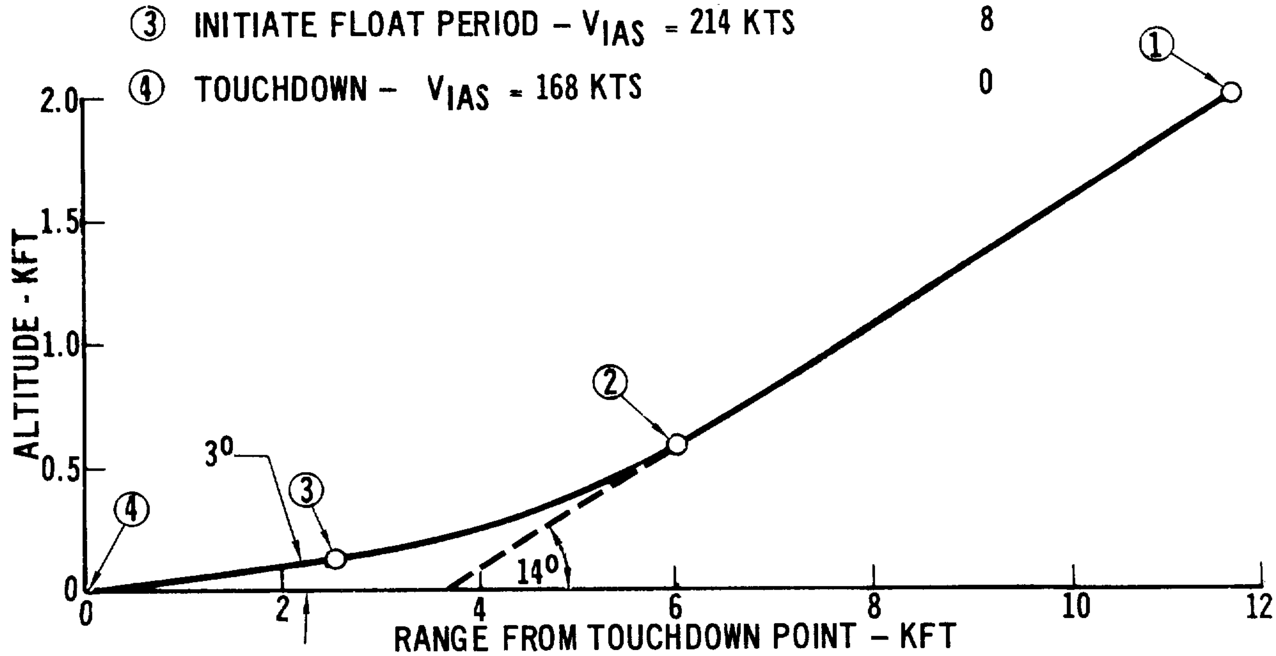

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Figure 3–97

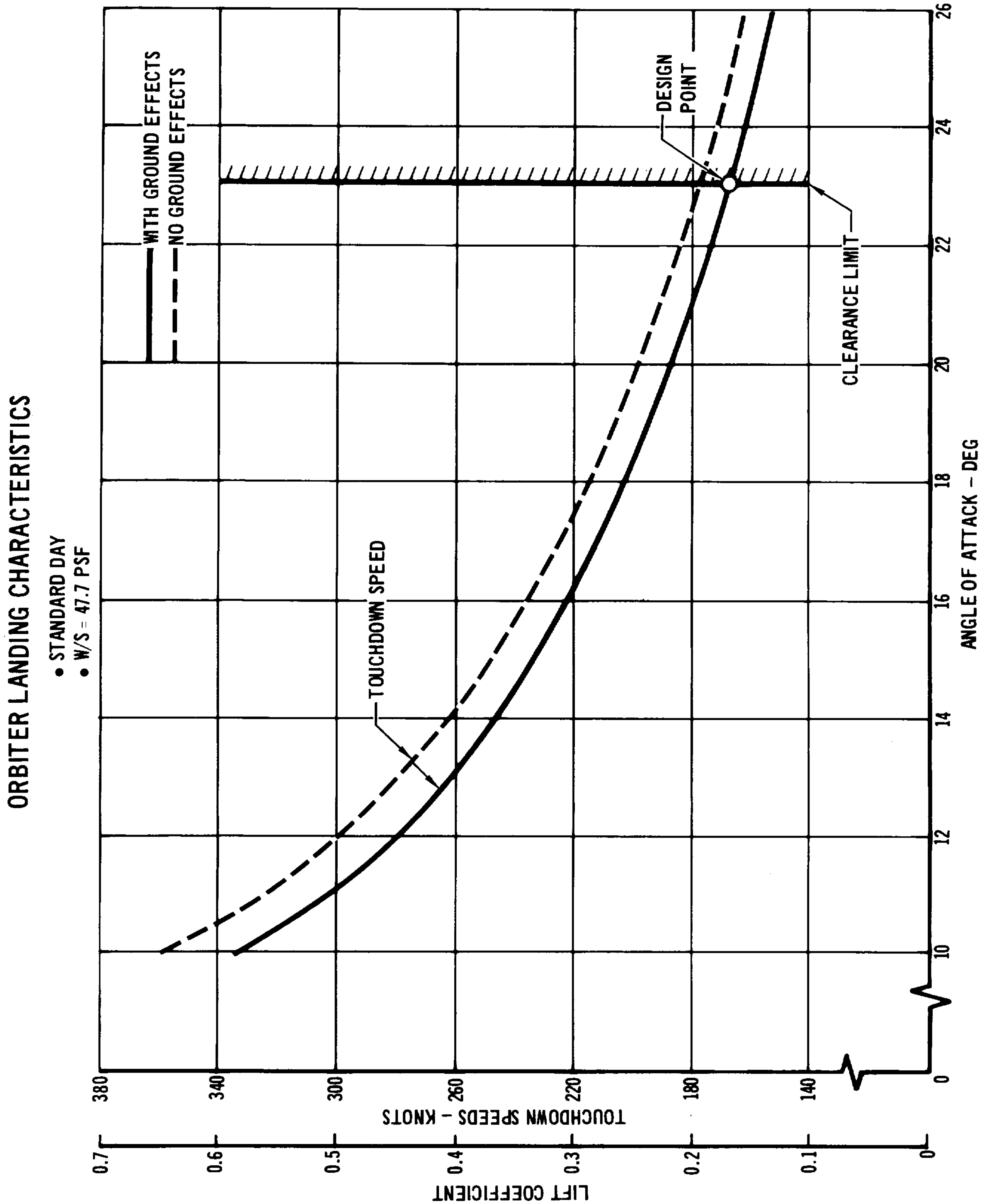
ORBITER LANDING CHARACTERISTICS
• STANDARD DAY
• W/S = 47.7 PSF
WITH GROUND EFFECTS
NO GROUND EFFECTS
DESIGN POINT
CLEARANCE LIMIT
TOUCHDOWN SPEED
ANGLE OF ATTACK - DEG
0
10
12
14
16
18
20
22
24
26
TOUCHDOWN SPEEDS - KNOTS
380
340
300
260
220
180
140
LIFT COEFFICIENT
0.7
0.6
0.5
0.4
0.3
0.2
0.1
0

Figure 3-98

CARRIER IFR APPROACH

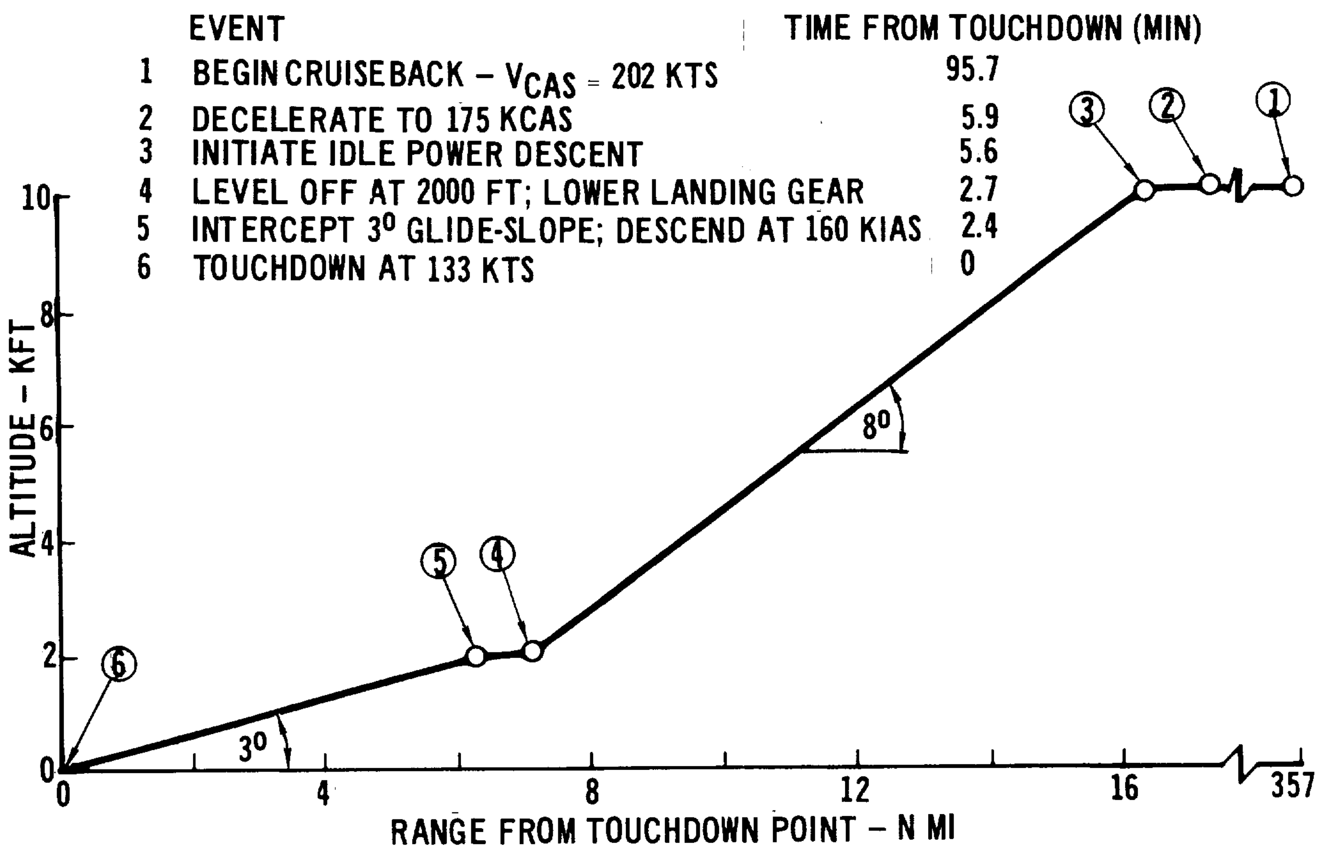

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Figure 3–99

3.4.5 Carrier Go-Around - The same philosophy that was used to select the baseline Orbiter go-around system applies to the Carrier, i.e., the baseline Carrier go-around system has the capability to climb-out to a pattern altitude of 2000 feet following a wave-off at 50 feet altitude, and make an IFR approach from an outer marker located 6.3 na mi from the end of the runway.

The go-around pattern geometry and corresponding sequence of events are shown in Figure 3-100. The rate of climb and altitude time histories are presented in Figure 3-101. Figure 3-102 presents the time histories of thrust and fuel expended during the Carrier go-around. Based upon a Carrier weight of 485,209 lbs. and a specific fuel consumption of 0.424 lb. of propellant per lb. of thrust per hour, a total fuel weight of 4000 lb. is required to execute a go-around pattern.

3.4.6 Carrier Landing - The Carrier landing characteristics are shown in Figure 3-103. The estimated ground effect produces a twenty-five percent increase in the lift-curve slope which results in a touchdown speed of 135 knots at the design angle of attack of 12 degrees for the standard, sea-level condition. A wing loading of 38.5 lbs/ft^2 was assumed which corresponds to a touchdown weight of 462,170 lbs. If all of the go-around and contingency fuel is expended, the landing weight is reduced to 450,940 lbs., resulting in a touchdown speed of 133 knots at 12 degrees angle of attack.

3.4.7 Automatic Powered Landing - A digital computer simulation of the terminal approach and landing phase with automatic control was performed for both the Orbiter and Carrier vehicles. Events occurring prior to the initiation point of this simulation for the Orbiter include (1) arriving over the landing site at a hight altitude, (2) spiraling down while remaining within the range of an unpowered landing if needed, and (3) deploying, starting, and checking out the engines. Following these events, the Orbiter is aligned with the runway and stable level flight at the approach altitude and speed is established. At this point, the simulation commences. The Carrier, which has been under powered flight for a significant time, has only to establish stable level flight at its approach altitude and speed and align with the runway.

A block diagram of this simulation is shown in Figure 3-104. The significant features of this simulation are: (1) The thrust is commanded to zero at the initiation of flare, (2) the pitch autopilot has a constant transfer function, and

CARRIER GO-AROUND PATTERN SEQUENCE OF EVENTS

EVENT	TIME AT INITIATION OF EVENT REFERENCED TO WAVE OFF, SEC.
① CLIMB OUT AT $\dot{h}$ = 33.3 FT/SEC; γ = 5.77°	0
② 20° BANKED TURN; $\dot{h}$ = 30 FT/SEC; γ = 5.22°	30
③ LEVEL FLIGHT 20° BANKED TURN	63
④ CRUISE AT h = 2,000 FT – V_{CAS} = 191 KTS	119
⑤ LEVEL FLIGHT 20° BANKED TURN	280
⑥ CONSTANT ALTITUDE DECELERATION	369
⑦ DESCEND ON 3° GLIDE SLOPE; $\dot{h}$ = 15.5 FT/SEC	386
⑧ TOUCHDOWN AT 133 KTS	514

CARRIER GO-AROUND PATTERN

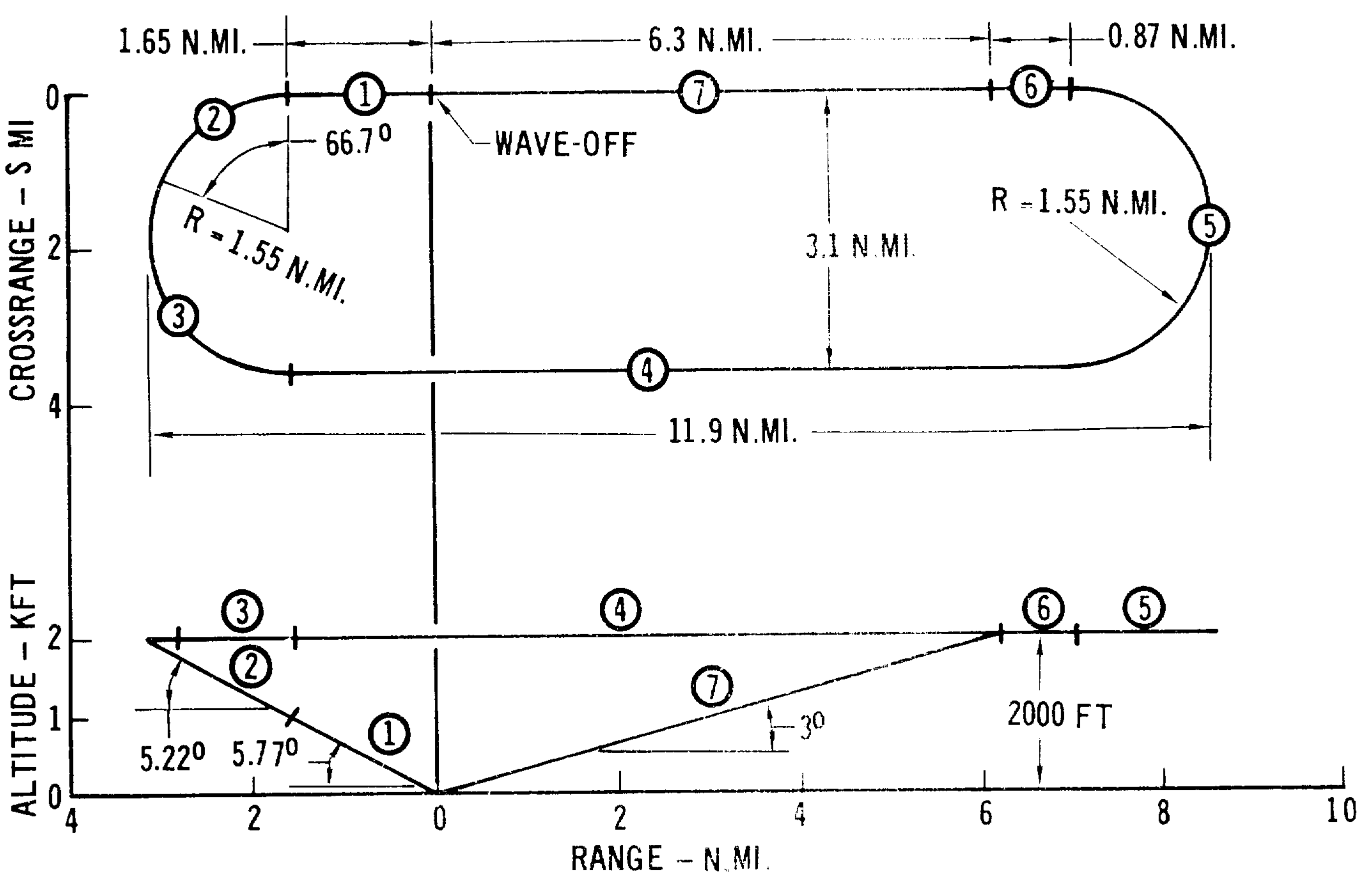


Figure 3-100

CARRIER GO-AROUND ALTITUDE AND RATE OF CLIMB TIME HISTORY

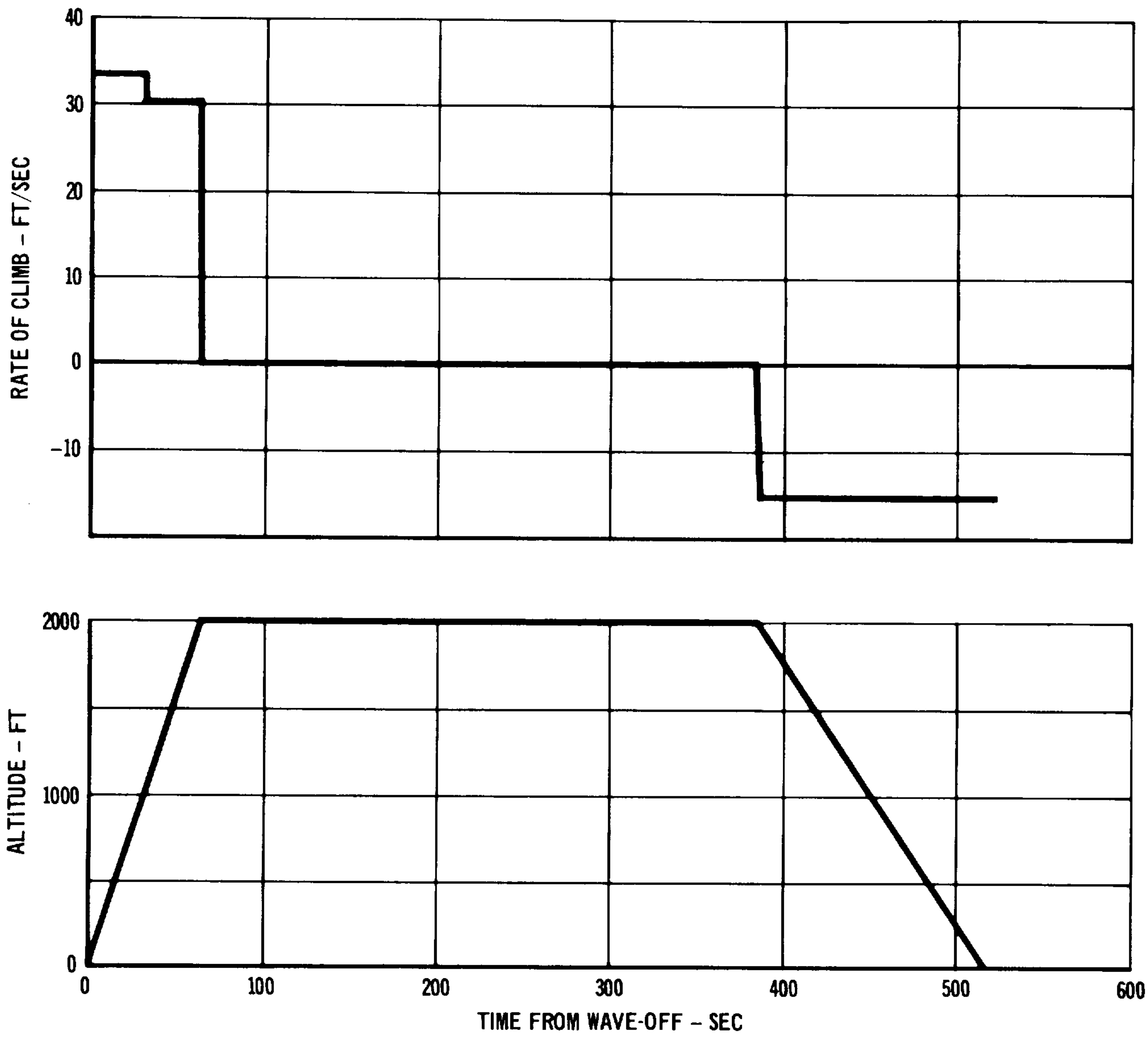


Figure 3-101

CARRIER GO-AROUND THRUST AND FUEL REQUIREMENTS

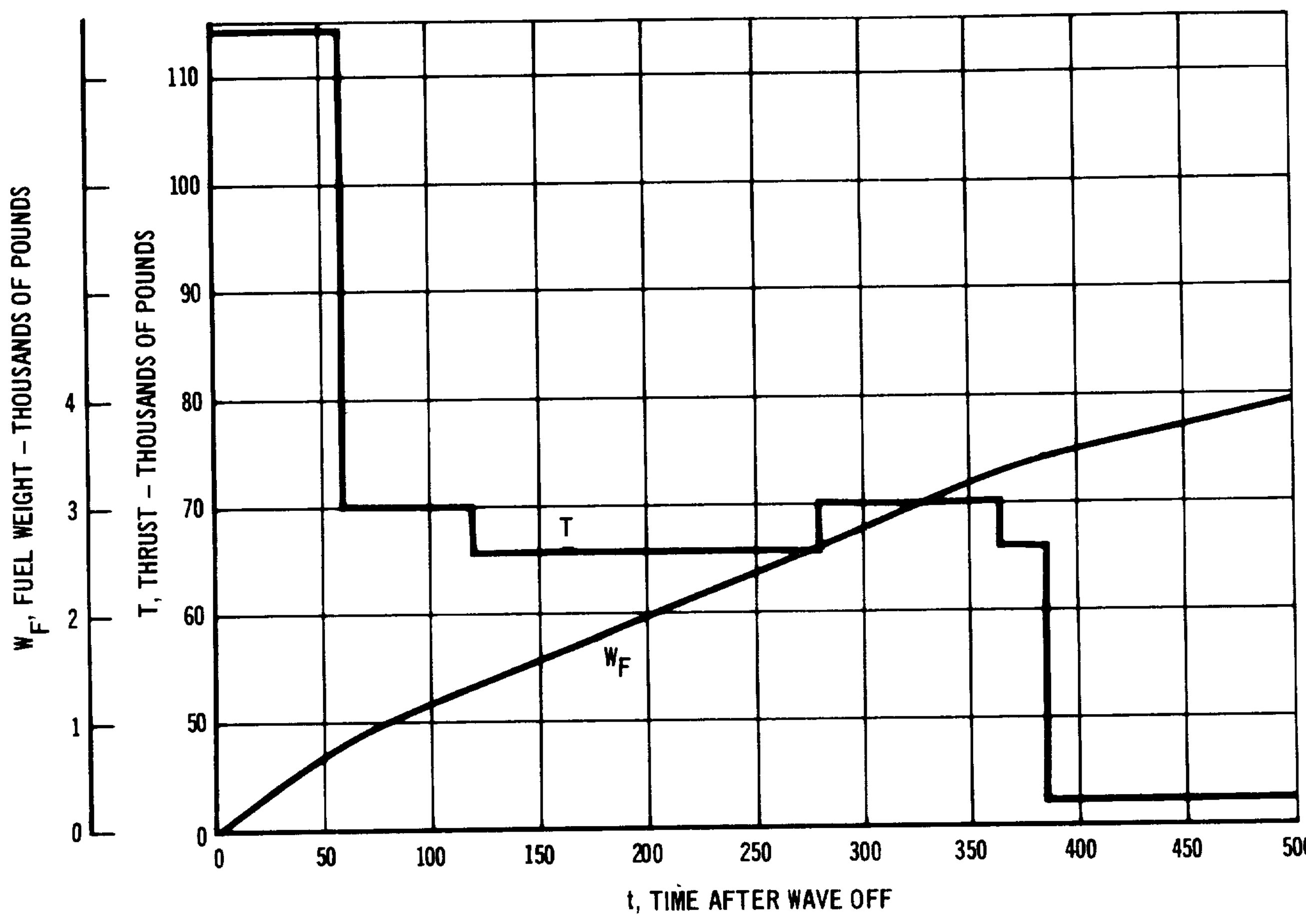


Figure 3–102

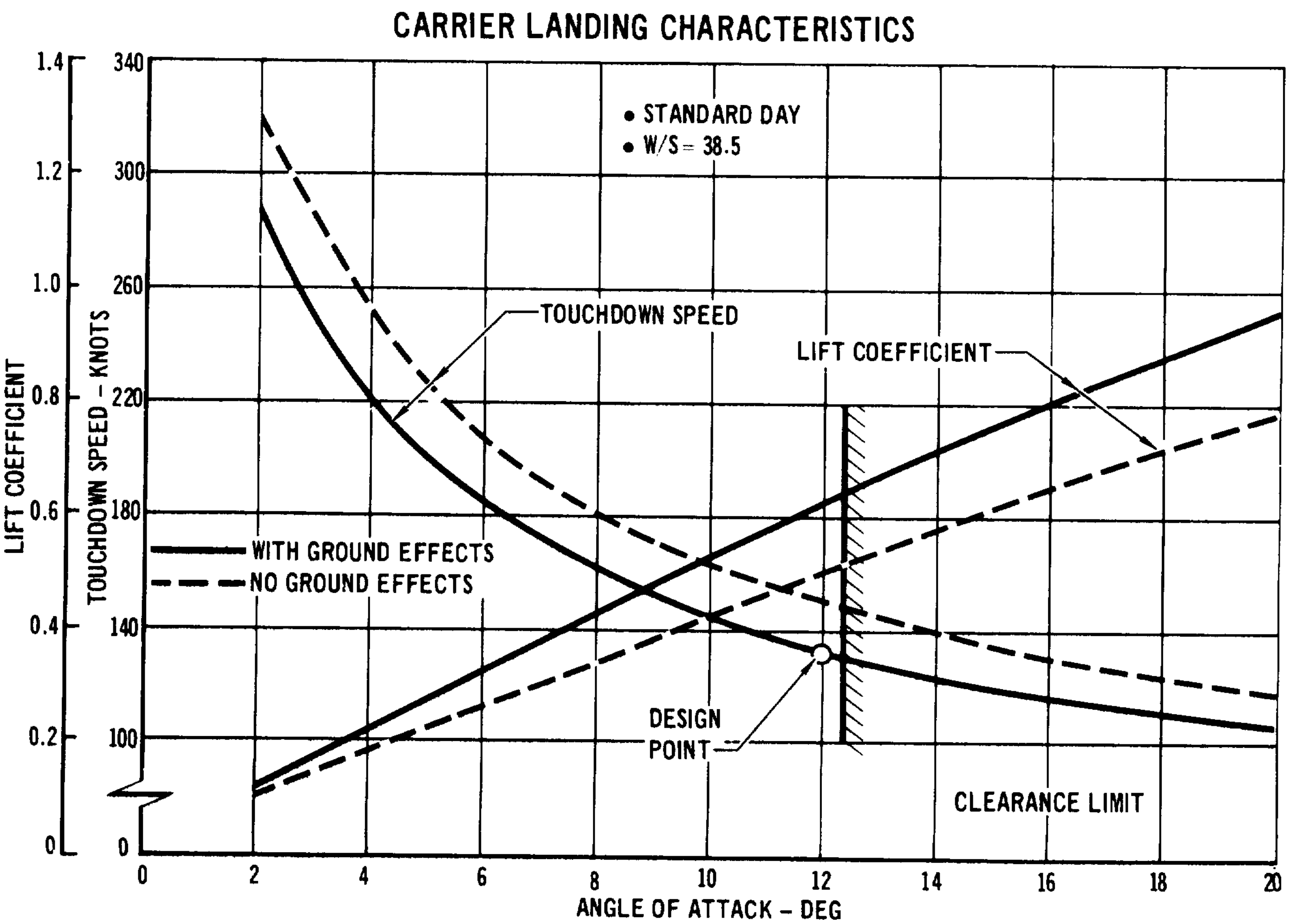
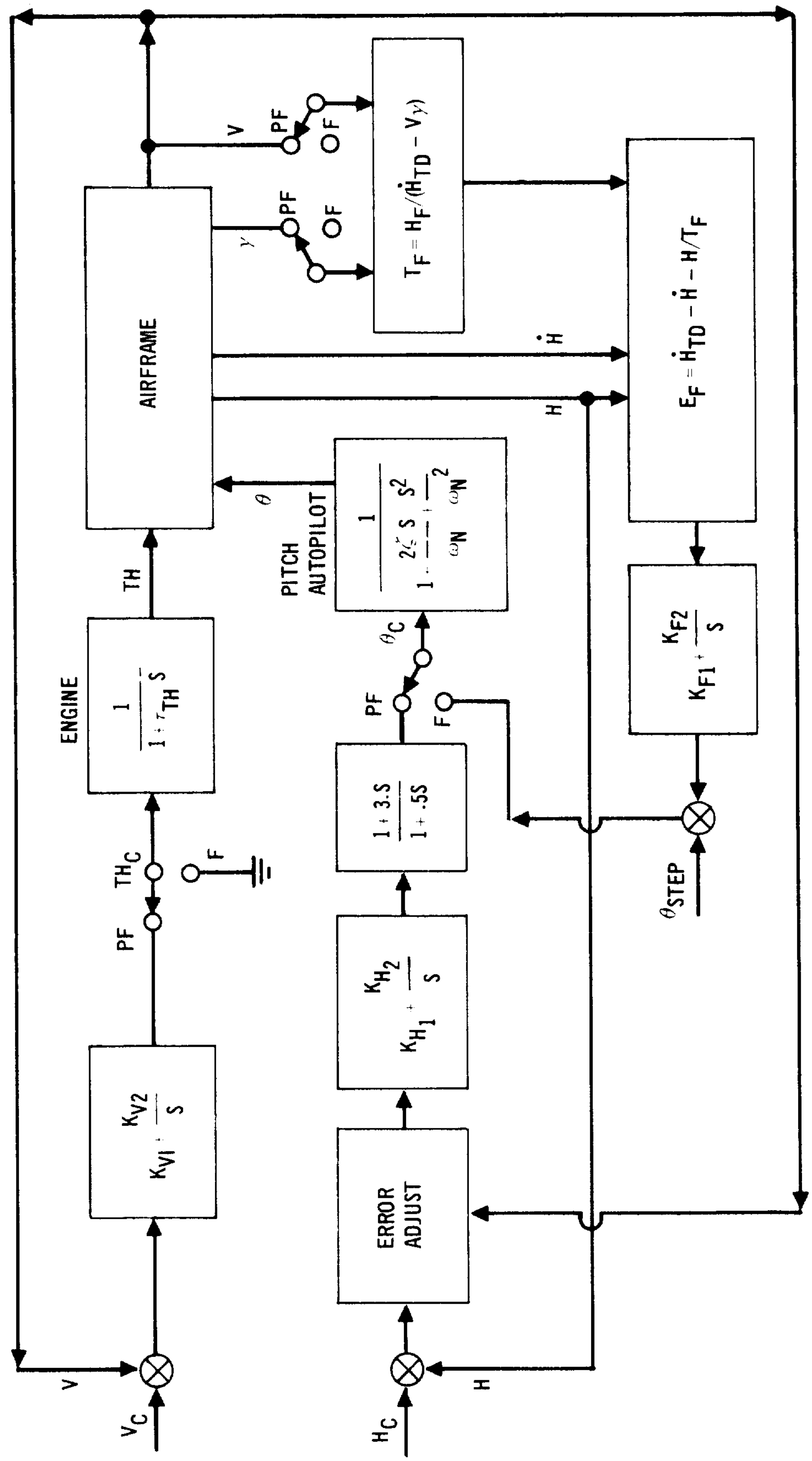

CARRIER LANDING CHARACTERISTICS
• STANDARD DAY
• W/S = 38.5
TOUCHDOWN SPEED
LIFT COEFFICIENT
WITH GROUND EFFECTS
NO GROUND EFFECTS
DESIGN POINT
CLEARANCE LIMIT
LIFT COEFFICIENT
TOUCHDOWN SPEED - KNOTS
ANGLE OF ATTACK - DEG
1.4
1.2
1.0
0.8
0.6
0.4
0.2
0
340
300
260
220
180
140
100
0
0
2
4
6
8
10
12
14
16
18
20

Figure 3-103

AUTOMATIC POWERED LANDING
Simulation Block Diagram



F - FLARE PF - PREFLARE E_F - EXPONENTIAL FLARE T_F - FLARE TIME

Figure 3-104

(3) all necessary ground-generated data, such as the glide slope determination, is available. The simulation is completed at touchdown. The pertinent parameters associated with the simulation are shown in Table 3-5. Only the vertical plane was investigated in this study but prior simulations which included the lateral plane demonstrated that a satisfactory response was easily obtained when the vertical plane response was satisfactory.

Both the Orbiter and Carrier vehicles start at an altitude of 2000 feet at a down range of 7 to 8 nautical miles. When the 3 degree glide slope to the touchdown point is intercepted, the landing guidance system commands the vehicle to pitch down to achieve this slope. The airspeed at the start of the simulation is commanded to decrease to the airspeed desired at the initiation of flare. This slowdown is accomplished shortly after the glide slope is intercepted. By the time the flare maneuver occurs, the flight path and velocity are well established with the transients adequately damped.

The results of the simulations for the Orbiter are shown in Figure 3-105 and 3-106. The nominal trajectory shows that the transients are damped out by the time that flare occurs. The flare details show a smooth landing after the 50 foot altitude flare initiation. The touchdown conditions are a sink rate of 3.5 feet/second, airspeed of 182 knots, angle of attack of 22.0 degrees, and a pitch attitude of 21.4 degrees based on a wing loading, of 49.4 lb/ft^2 and without considering ground effects. In as much as this W/S differs slightly from that quoted in other parts of the report due to design iteration, the corresponding touchdown velocity is somewhat higher also. The flare maneuver takes 4.8 seconds.

The resulting nominal trajectory for the carrier is shown in Figures 3-107 and 3-108. Again the transients have been damped prior to flare. The flare maneuver starts at an altitude of 30 feet and produces a smooth landing. The touchdown conditions are a sink rate of 3.0 feet/second, airspeed of 136 knots, an angle of attack of 11.9 degrees, and a pitch attitude of 11.1 degrees. The flare maneuver requires 3.8 seconds. The pitch attitude limit constraint is not exceeded.

The equipment required to perform these terminal landings is within the capabilities of currently available equipment. Air data instruments of a conventional type to provide an indication of airspeed and altitude is adequate. An updated inertial navigation system could be used instead of the air data altimeter.

A low range radar altimeter with a 500 foot maximum range and altitude error of less than 2 feet in the vicinity of the flare initiation is required to perform the flare maneuver. The autopilot and flare electronics are parts of the existing computer. The establishment of the 3 degree glide slope can be done by an AILS or a precise ground based radar. For a more detailed discussion of the avionic equipment required to implement the automatic landing capability, see Section 4.3.5 in Volume I, Part I of this report.

Table 3-5

PARAMETERS FOR AUTOMATIC POWERED LANDING SIMULATION

PARAMETER	ORBITER	CARRIER	UNITS
K_{V1}	960.	1500.	Lbs/Knot
K_{V2}	48.	100.	Lbs/Knot Sec.
K_{H1}	.015	.030	Deg/Ft.
K_{H2}	.001	.002	Deg/Ft. Sec.
K_{F1}	.6	.69	Deg/Ft.
K_{F2}	.22	.069	Deg/Ft.
TH	2.	.5	Sec.
W_N	1.25	1.25	Rad/Sec.
	.7	.7	--
H_{TD}	3.	3.	Ft. Sec.
H_F	50.	30.	Ft.
STEP	3.5	2.65	Deg.
Initial Velocity	222.	180.	Knots
Velocity at Flare	200.	145.	Knots

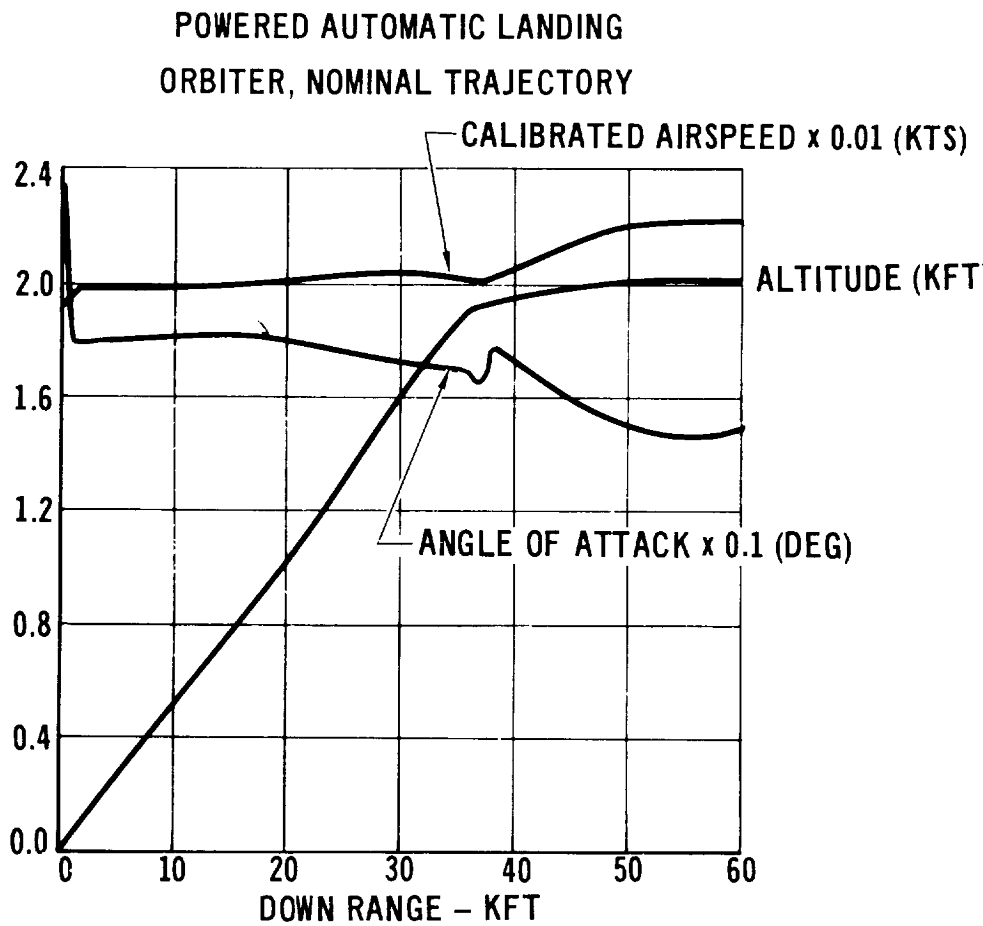


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Figure 3-105

POWER AUTOMATIC LANDING
ORBITER, FLARE DETAILS

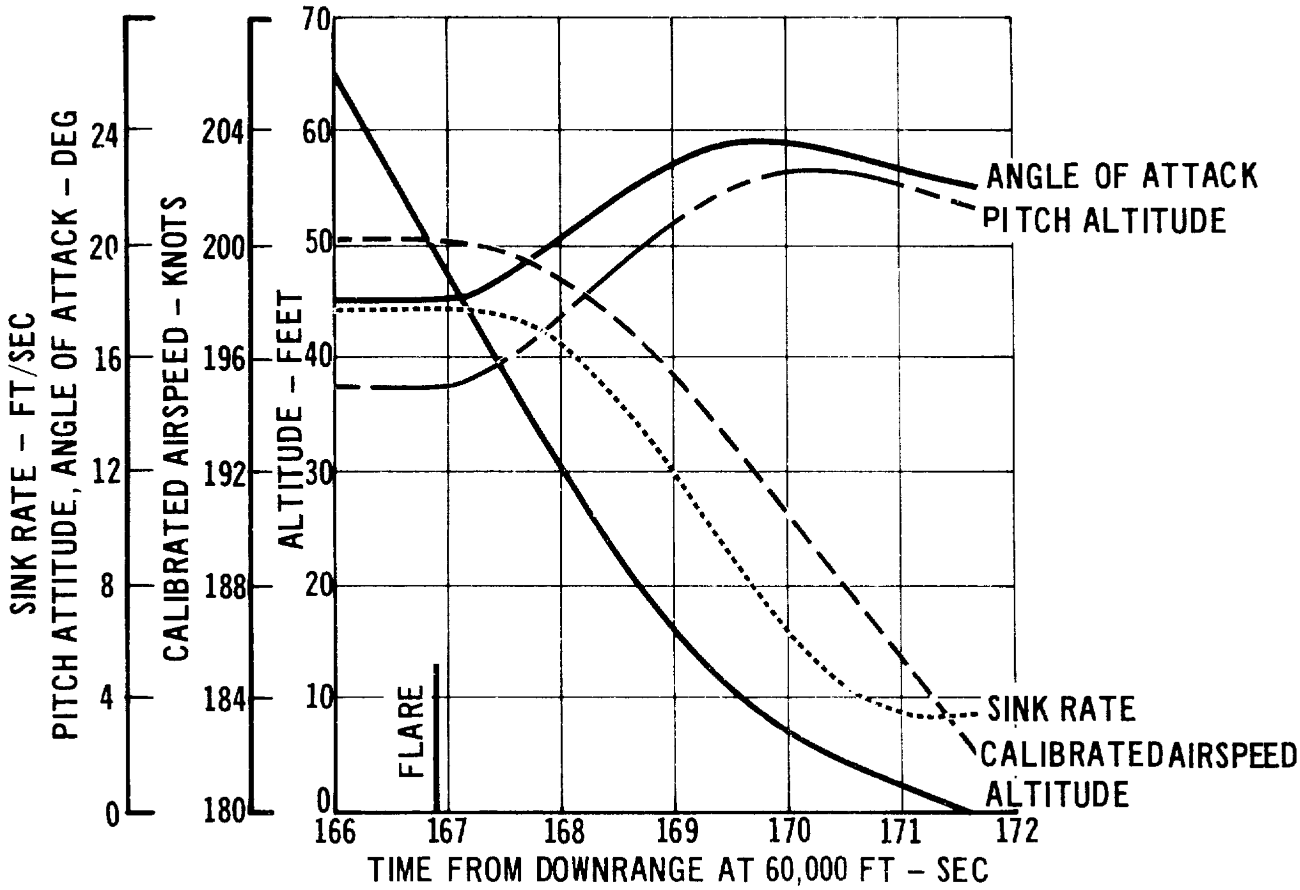

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Figure 3-106

POWERED AUTOMATIC LANDING
CARRIER, NOMINAL TRAJECTORY

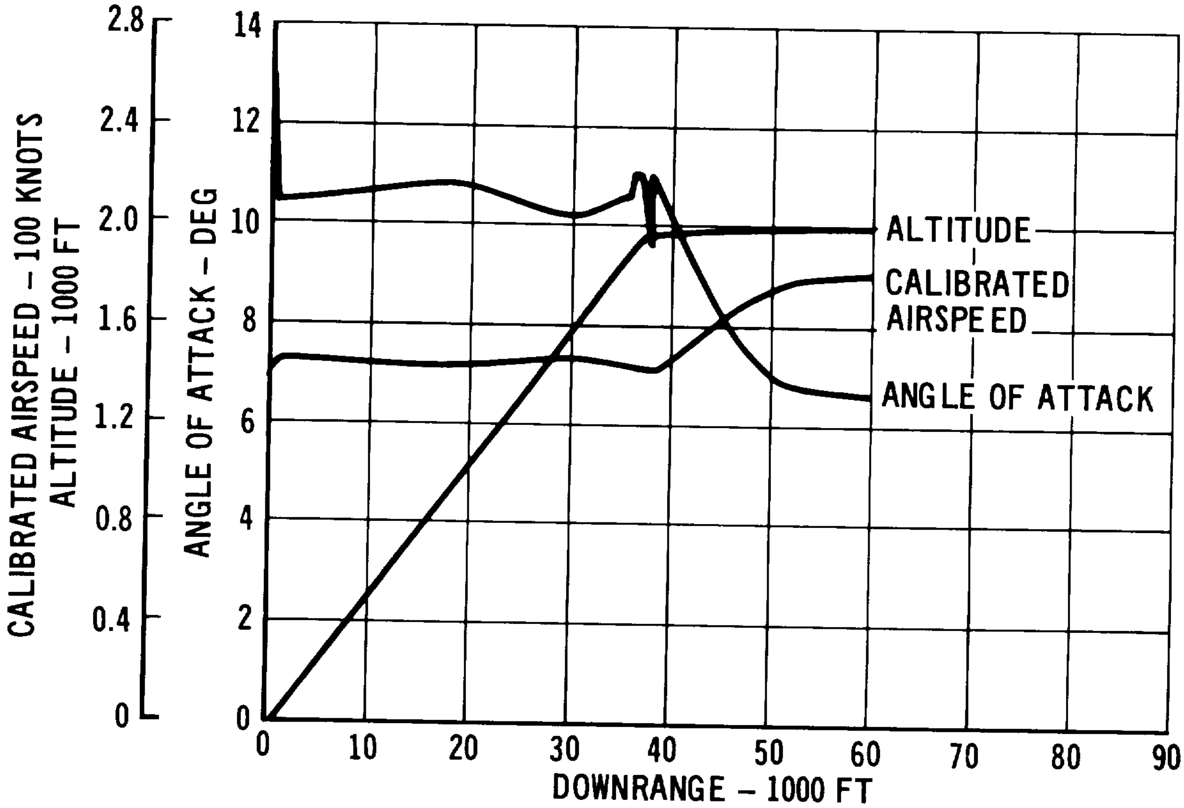

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Figure 3-107

POWERED AUTOMATIC LANDING
CARRIER, FLARE DETAILS

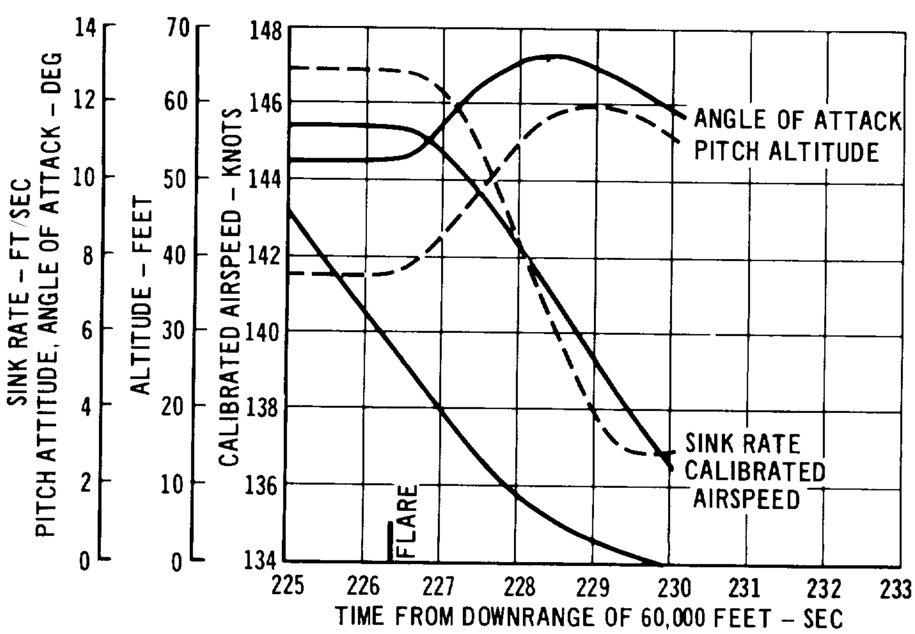


Figure 3-108

3.5 Abort Performance - An abort in its broadest sense is an interruption to planned or scheduled events or activities. An abort is required whenever a sequence of events or activities proceeds in a manner such that danger to personnel or equipment will ensure unless the sequence is interrupted or terminated. Abort plans and procedures attempt to do three things:

- o Prevent injury to personnel
- o Reduce the amount of damage to equipment and facilities;
- o Reduce the time loss attendant upon the abort activities.

Using these definitions and goals abort plans and procedures for the baseline system where outlined. Abort analyses including trajectories and aero-thermo considerations were conducted for the baseline mission and corresponding implications for vehicle design were identified. In this analysis, emphasis was placed on investigating the possible causative factors leading to an abort and determining a feasible approach to the overall problem.

3.5.1 Abort Philosophy - In the airline industry abort procedures and operations are based on the underlying assumption that the crew and passengers are committed to the safety and integrity of the airframes; whatever happens to the airframe also happens to them. This philosophy is reflected in the painstaking inspection and certifications of the aircrews, groundcrews, flight vehicles and support facilities to insure the safety and reliability of men and machines. This philosophy recognizes that aborts are inherently probable, and abort procedures and responsibilities are defined by the FAA and the airlines. These procedures include ground simulations and in-flight tests of aircrews and vehicles which include known and expected anomalies that would require an abort during the flight operations. This philosophy is to be carried over to the ILRVS Program.

It is recognized that the airline philosophy governing aborts is based on aircraft experience, vehicle availability and final goals. For the airlines the goal is to get the customer from point A to point B as quickly and conveniently as possible. However, in the case of the ILRVS program, getting to point B may be more than a matter of convenience; it may be a matter of survival when point B is the Space Station/Space Base complex. The possible loss of the timely arrival of the cargo and passengers at their destination may be of great import in the long-term operations of the space program.

Commerical airline companies number their passengers in the millions, and their daily operations average 1 or 2 hours between take-off and landings. For the ILRVS based on a 100 man space base with 120 day rotation rate for all personnel, this amounts to a total passenger loading of 300 men per year. Whereas it is not feasible to provide escape and abort training for millions of passengers, it would appear to be entirely feasible to provide such training for 300 persons per year. Furthermore, the shuttle vehicle will provide very sophisticated life support systems for the passengers and crew, and the average flight time between take-off and landing will probably exceed 24 hours, and may run as long as 7 days, which is a quite different situation than is encountered by the average air traveler. Therefore, for the ILRVS, it was assumed that each passenger will receive some degree of training and indoctrination beyond the level presently provided in over-ocean flights by commercial airlines, but certainly not more complicated than parachute training.

3.5.2 Operational Phases and Types of Aborts - Regularly scheduled airlines schedule maintenance operations along with flight operations and often work three shifts/day, seven days/week in order to maximize the utilization of the aircraft and to minimize downtime and idle periods. This means that for all practical purposes, the entire lifetime of the vehicle is accounted for until its final flight. As far as the airlines are concerned, any abort is serious, but a maintenance abort is conceded to be less serious than a flight abort. This is because a maintenance abort does not involve the exposure of personnel to serious injuries, and usually does not involve the loss of the vehicle, whereas both of these conditions attend a flight abort. Table 3-6 provides a side-by-side and phase-by-phase comparison of airline operations and ILRVS operations and the type of abort which would be encountered during each. There are some features of the ILRVS vehicle which will invoke new and different operating techniques (such as the vertical launch procedure and the separation procedure) for in-flight aborts, but it is expected that maintenance aborts will be essentially the same. The impact of an abort on the overall mission will also require that a slightly different philosophy be developed, because if a scheduled logistics shuttle is delayed too long, the totality of this impact must include an estimate of the effect on the space station in addition to the effect on the shuttle itself. ILRVS operations should include some alternatives for such contingencies.

Table 3-6

ABORT MODE COMPARISONS

AIRCRAFT ABORT MODE	ILRVS ABORT MODE
Ground Checkout, Refurbishment & Refuel Prior to taking on passengers or cargo, the vehicle is carefully checked over by the groundcrew. Any "squawks" written up by the flight crew are cleared up, and the flight data is read out and analyzed in order to spot problems before they occur. An abort in this phase would require a replacement aircraft while the fault is corrected.	Ground Checkout and Refurbishment For the ILRVS, essentially the same procedures will be followed, but the depth of the investigations and the thoroughness of the data analyses will be greater and more complex by virtue of the size and complexity of the vehicle and the more demanding flight envelope. An abort would require a re-ordering of priorities and a re-shuffling of planned flight schedules and payloads, and would impact on Space Base operations, too. Recovery from the abort would be effected when the faulty vehicle is repaired and both ILRVS and Space Base operations have been restored to planned levels and schedules.
Preflight Checkout and Engine Tests Most of the aborts encountered in this phase are connected with the engine tests. If it should become necessary to replace an engine, the scheduled flight is usually aborted unless a substitute aircraft is available. Scheduled airlines can usually recovery from these aborts by pooling passengers with other airlines, leasing another aircraft, or transporting passengers by bus or train, where such surface transportation exists.	Preflight Checkout and Engine Tests For the ILRVS, this phase is almost a continuation of the previous phase, and, as with the airlines, the major problems are expected to be in connection with the engines. However, for the ILRVS, recovery from an engine problem is going to be more difficult. The impact of an abort in this phase will be the same as stated above. Recovery from abort will also be similar. Modularization of the payload modules and interchangable vehicles should minimize schedule fluctuations due to the abort of one carrier or one orbiter. Loss of both vehicles would be more serious, and recovery from this situation is more difficult.
Load and Embark; Taxi to Runway By the time an aircraft has reached this phase, there should be a high level of confidence in the vehicle, and only a real emergency would lead to an aborted flight. For example, a recent flight was aborted in this phase when a coffeepot short-circuit indicated a possible fire. The flight crew aborted to insure the safety of the vehicle and the passengers.	Transported to Launch Site; Erected, Mated, Fueled & Loaded An abort during this phase would be unlikely, and would probably be associated with the AGE rather than with the flight vehicles. Recovery from that type or abort would be easier, and would probably only involve a time delay. However, damage to the flight vehicles could occur, for example, if the erection/transportation devices should fail catastrophically. In this case, the abort would be very serious, and recovery would be difficult.

Table 3-6

(Continued) ABORT MODE COMPARISONS

AIRCRAFT ABORT MODES	ILRVS ABORT MODES
<u>Landing, taxi and De-planing</u> A normal abort from the landing phase of airline operations usually consists of a wave-off and a go-around. These aborts are intact aborts. If an aircraft encounters the ground too soon or too late, some degree of structural damage is incurred. Passengers and crew are committed to the integrity of the vehicle, and landing sites are equipped to cope with crashes and over-runs. Runway barriers are employed increasingly to cope with overruns, which improve the probability of an intact abort from that situation. Little can be done if the aircraft crashes on the approach leg, due to the high speed and slow response characteristics of the vehicle. Even after a successful touchdown, some anomalies could abort the landing phase. For example, if the aircraft thrust reversers fail, the run-out distance could result in damage to the aircraft, and emergency procedures for getting passengers and crew out of the vehicle are well defined, with information available to everyone.	<u>Landing and Shutdown, Debarking</u> Both the Carrier and the Orbiter segments of the ILRVS are designed to perform the go-around manuever, so in this respect they are very similar to regular aircraft. However, after landing, it is very likely that little or no "taxiing" will be done, due to the large size of both vehicles and the extensive fuel requirements needed for such manuevers. Also, safing operations will be necessary for both segments, due to the hazardous fuels used and the complexity of the on-board power and life-support systems. For the orbiter, some type of 'quarantine' facility may be needed for returning personnel and experimental animals. Aborts during this phase of the operations would probably require some means for smothering fires in the residual fuel systems, and a transportation/controlled environment shelter system, with provisions for extracting the personnel module and the crew from an unsafe vehicle.
<u>Turnaround</u> During this phase, aborts may be encountered due to the type of operations inherent in the servicing of these vehicles. Electrical power from service units, static electricity, de-icing fluids, fuel, tires and brakes, engine tests, and cockpit checks could lead to serious problems. Careful attention to details and procedures helps, but cannot insure against accidents during this phase. Aborts could result in loss of equipment and facilities, as well as personnel	<u>Turnaround</u> The designed-in reusable systems of the Carrier and Orbiter should minimize aborts during the turnaround phase of ILRVS operations. The extensive self-test and on-board checkout capability may even improve turnaround times. Structural inspections are expected to be time-consuming, and will be serious sources of aborts, since the flight regime is such an unknown factor. Every effort will be made in designing these vehicles for integrity, but until experience proves the validity of the design, it is to be expected that caution will prevail; "When in doubt, check it out," and some vehicles will be unnecessarily impounded in the name of safety. Operations should be scheduled to recover from these aborts as smoothly as possible.

Table 3-6

(Continued) ABORT MODE COMPARISONS

AIRCRAFT ABORT MODE	ILRVS ABORT MODE
Takeoff and Initial Climb to Altitude The transition from landborne to airborne operations is a very critical phase. Aborts at this time are very hazardous to personnel and equipment. If the aircraft undergoes a structural failure, the crew and passengers are in great jeopardy. Successful aborts are measured in terms of human lives; as the saying goes, "Any landing you can walk away from is a good one." Loss of equipment is considered secondary, although every effort should be made to save it.	Ignition, Liftoff, Ascent, Staging and Insertion Aborts before lift-off are hazardous in that hot engine systems and hazardous fuels are in close proximity, and catastrophic fires could occur. Abort systems for this phase could include contingency measures adjacent to the launch site. Aborts after liftoff are very hazardous to crew and passengers, especially under the series-burn operations. Until the orbiter has passed the 2000 ft altitude, it is virtually impossible for the orbiter to perform an intact abort. After this altitude, it is possible for the orbiter to start its engines and stagger into an abort trajectory followed by an early landing. After staging, an abort to orbit may be preferable to flight termination, depending on launch site and environmental constraints, and also the reason for the abort.
Cruise An inflight abort is occassionally necessary, but unless there has been a catastrophic structural failure, chances for a successful abort are good. The aircrew will callup alternate landing sites and invoke emergency procedures in accordance with the established regulations.	Orbit Aircraft pilots faced with landing problems often wish they could extend their flight time indefinitely. Now, it will be possible to abort in orbit until assistance can be provided to remedy the situation. New abort methods will have to be developed, based on expected or possible assistance from the Space Station/Space Base, or from the next ILRVS mission. Chances for successful aborts in this phase would appear to be very good.
Descent from Cruise An abort from the descent phase is usually due to an equipment malfunction, which in turn would jeopardize a successful completion of the next phase. Any other abort mode during the descent phase would be quite similar to a cuise phase abort. Mechanical anomalies during descent are common factors in crew certification flights.	Retrograde and Re-Entry; Cruise and Descent These operations are common to the carrier and the orbiter segments of the ILRVS. These phases are more critical than the descent phase of commercial airlines, and the aborts would be correspondingly more difficult. In the event that the carrier could not manuever during and after staging, carrier recovery operations would be affected, but the carrier would still be able to perform an intact abort. The same is true for the orbiter, except that the orbiter can also abort to orbit, and await rescue operations from the Space Station/Space Base, or from the ground.

In summary, there are two types of aborts; those occurring during scheduled maintenance operations, and aborts occurring during normal flight operations. Flight aborts are more serious than maintenance aborts because of the possible loss of lives and serious damage to or loss of the vehicle.

3.5.3 Maintenance Abort - A maintenance abort is different from a "turnaround" abort. Maintenance operations are clearly established during the design and introduction of an aircraft into airline operational service. For the ILRVS, the program is not quite so clear because the shuttle vehicle undergoes considerably more recertification than is the case of commercial aircraft. The more extensive refurbishment is required by the very long duration of flight operations as compared to regularly scheduled airlines. Furthermore, the airlines have more than 100 flight vehicles, and it is usually possible to acquire other vehicles on short notice in order to recover from a maintenance abort, but it is not likely that the ILRVS can be operated this way. If the shuttle vehicle undergoes a maintenance abort, a domino effect will occur and priorities must be reordered so as to recover from the abort as rapidly as possible, either by increasing the duty cycle of the remaining vehicles, or by reducing support for or operation of the Space Base until the deficiencies can be overcome.

In summary, maintenance aborts for the ILRVS are much more serious than are maintenance aborts for the scheduled airlines. This is due primarily to the effect on the Space Station or Base if a scheduled flight is not completed due to a maintenance abort, or for any other reason.

3.5.4 Flight Aborts - For the ILRVS, flight aborts are defined in much the same way that they are defined by the airlines. That is, a flight abort is called for at any time at the discretion of the command pilot. This situation will also hold for the ILRVS with the command pilot having the same responsibilities. In the case of the ILRVS, the pilot will have a great deal of information upon which to base an abort decision and operation because of the high degree of on-board diagnostic capability "built-in" and the "fail operational" design philosophy followed during the development phase. The pilot will still have the responsibility for an abort decision, and it is likely that the same approach to abort operations will prevail; that is, aborts will be simulated during crew training activities and in-flight training phases. The degree to which aborts will be simulated during in-flight crew certification has not yet been established. The risk involved must be carefully weighed against the advantages of such procedures before extensive real-time abort activities are undertaken.

During the development phase, when the vehicle and subsystems are being tested in planned, sequential steps, it is very probable that inadvertent aborts will be encountered. Each of these will be investigated in much the same way that aircraft anomalies are investigated, and an extrapolation will be made to determine the likelihood of such an event occurring during operations. This proven approach to anomaly analysis will produce improved understanding of, and appreciation for, the capabilities of the shuttle and its carrier. The horizontal take-off and landing "go-around" exercises should prove of particular importance in crew training and preparation for aborts.

3.5.5 Abort Systems - During the development phase, escape systems will be included as standard equipment for the test crews. However, it is recognized that the presently known escape systems are not directly applicable to the shuttle configuration. For example, escape rockets, such as used during Mercury and Apollo, assumed that the command module could be treated as an independent entity. This approach was also taken in the design of several military aircraft. Another method is to provide the crew with ejection seats, or for pad aborts, to provide a quick egress system. Both of these concepts assume a non-hazardous environment in the vicinity of the vehicle, which may or may not be the case. In any event, an escape system which is designed to separate the personnel from the vehicle is not in consonance with a multiple usage facility and re-usable vehicle operations. Again the airlines provide guidance, in that the crew and the passengers are completely dependent upon the safety and integrity of the vehicle. Utilizing this philosophy, only minimal escape capability need be provided for the ILRVS. The requirement that crew and passengers be provided with a shirtsleeve environment would also dictate against any elaborate escape system. However, the ILRVS orbiter and carrier vehicles will have some type of crew escape system during the development phase, and it is very likely that the carrier, at least, will continue to have such a system for the crew. The orbiter, too, could continue the escape system into the operations phase. It would then only be necessary to develop some type of escape or survival system for the personnel module. The most stringent requirement for such an escape or survival system would be that it must protect the passengers in the event of the catastrophic structural failure of the orbiter vehicle. Such a failure could occur during any phase of the mission. While it is conceded that escape systems can only be included at the expense of payload, we must recognize that the personnel flights in support of a 100 man space station/space base would probably be only one out of four (based on a 120 day stay period and a 10 passenger shuttle flight), and might be as few as one out of eight or ten, depending on the number of passengers which can be accommodated on the space station. Under these conditions, the payload penalty for the escape or survival system only affects one flight out of four and could be designed to be usable in the Space Station/Space Base complex as well, thereby permitting the system to function as a special purpose payload.

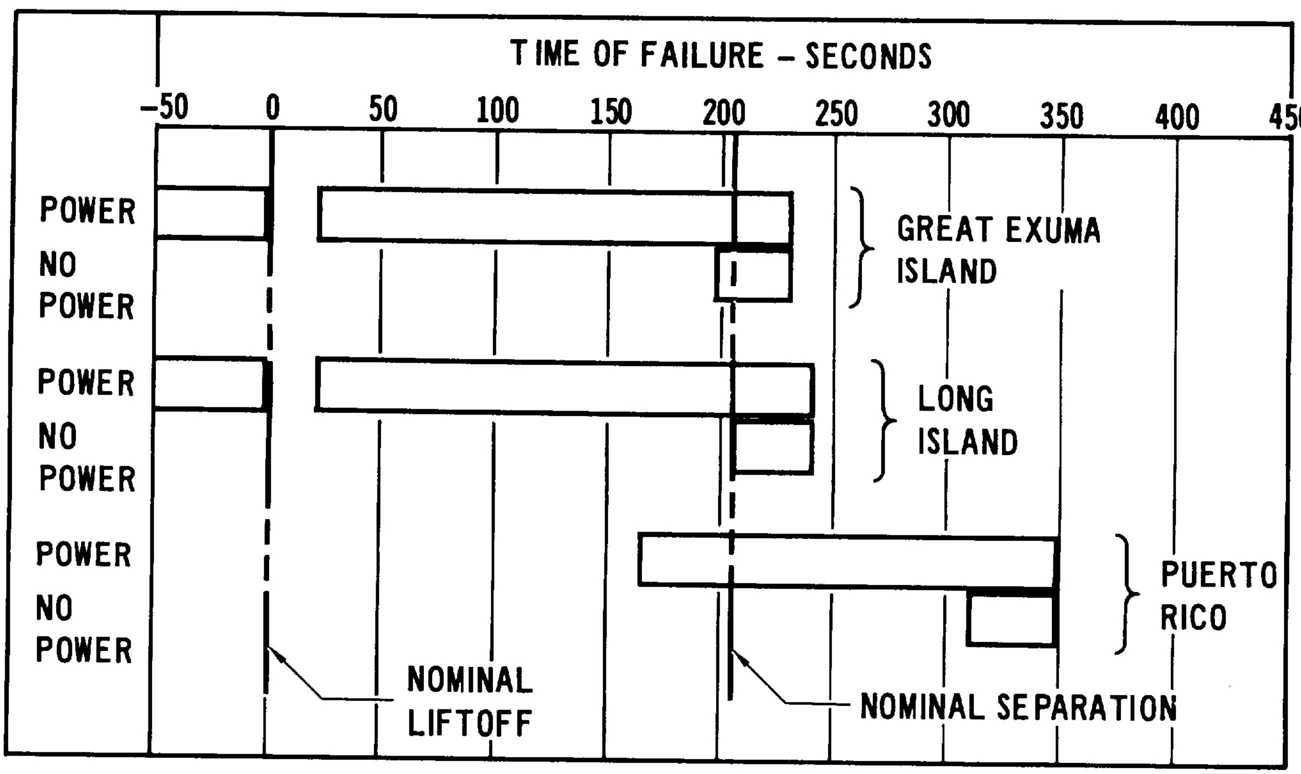
3.5.6 Implications for Design - A fundamental guideline employed in the design of the orbiter and carrier subsystem, was multiple redundancy. For avionics equipment, triple redundancy was implemented, i.e., the design philosophy was "fail - operational - fail operational - fail safe". For the mechanical equipment, a "fail operational - fail safe" design guideline was employed. This again reflects the current philosophy employed in the design of commercial aircraft. With this approach, the probability of an in-flight failure sequence which would result in an abort situation, is very small, and it is quite possible that providing launch abort capability is unwarranted. A more detailed discussion of this consideration is presented in Volume I, Part I. However, the design of the orbiter and carrier does permit safe aborts throughout most of the launch phase, and this capability does not incur exorbitant weight penalties. A discussion of the abort techniques is provided in part 3.5.7 of this section.

3.5.7 Abort Trajectories - Although by virtue of the system design (triple and double redundancy) the liklihood of an inflight abort is quite small, abort situations could arise through a sequence of multiple failures. The abort trajectory analysis was done for the case of multiple engine failure during launch. It was found that satisfactory reentries could be performed for failure at any time other than the first 20 seconds after lift-off. Trajectories calculated are applicable to any launch azimuth. However, in selecting landing sites for this study, special emphasis has been placed on a southerly launch into the nominal 55 degree inclination orbit. Figures 3-108 and 3-109 summarize the abort modes and choice of landing sites. It should be noted that a safe orbital abort for all azimuths is not possible for an ETR launch. This can only be achieved with a continental launch site. Such a launch site also has Carrier cruise range advantage, but the disadvantages of new facilities, air traffice interference, re-entry sonic boom, etc. probably outweigh the advantages.

Abort to Great Exuma Island: The nominal launch trajectory passes over Great Exuma Island approximately midway between Cape Kennedy and Haiti. From the ground track in Figure 3- 109 or the altitude-range profile in Figure 3-110 it can be seen that Great Exuma lies about 200 nautical miles down range from the nominal separation point. This location makes it a possible landing site in case of abort during much of the launch trajectory. If failure occurs prior to lift-off the Orbiter can under its own power lift off and fly to the nominal separation conditions using most of its propellant. From these conditions it can fly unpowered to Great

Exuma. Such a trajectory has been calculated. It's altitude-range profile is shown in Figure 3-110 and significant time histories are shown in Figures 3-111 and 3-112. The flight command was simply a constant 50 degrees angle of attack in an unbanked attitude. This required relaxing the load factor constraint from 3 to 4 g's, but no relaxation of heating constraints was necessary.

ORBITER ALTERNATIVE ABORT MODES



ILRVS-416F

Figure 3–108

ABORT LANDING SITES

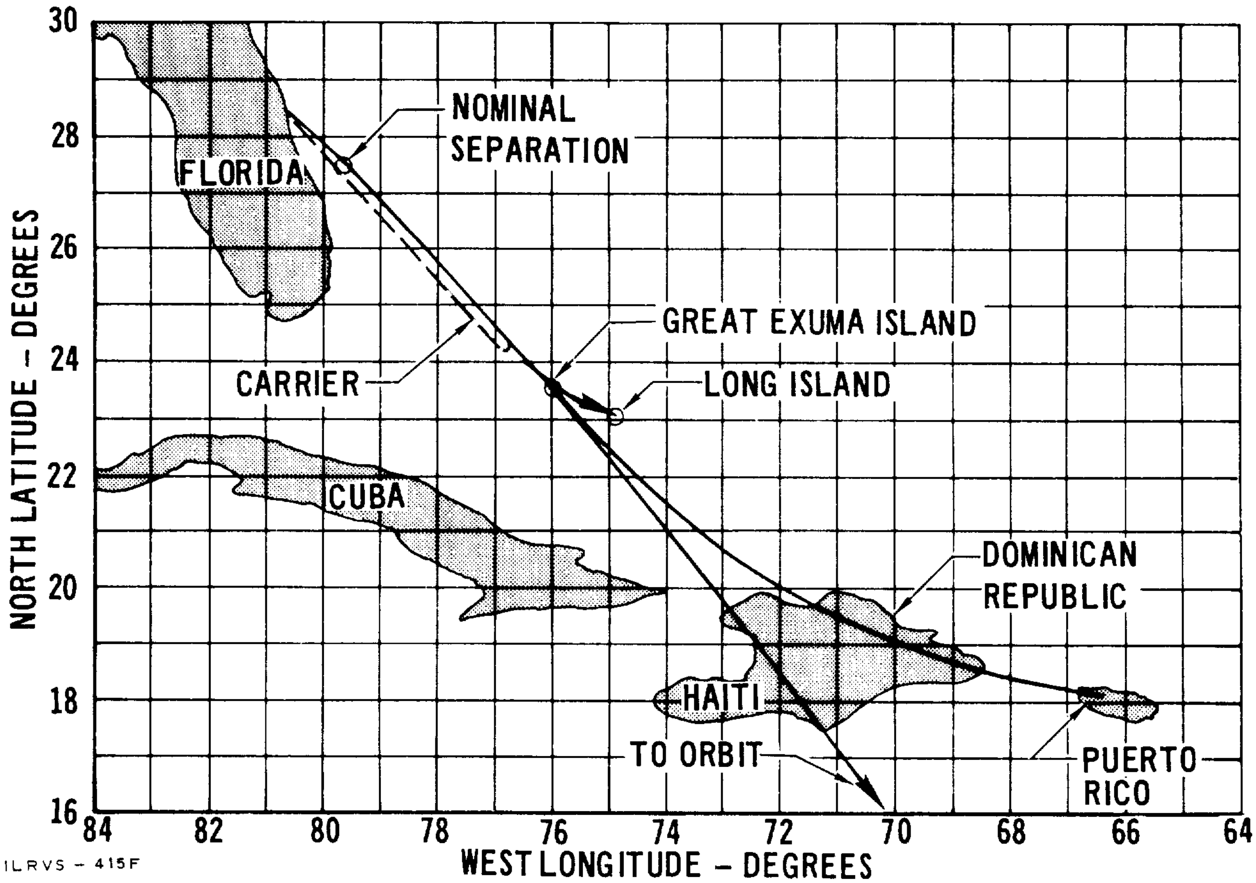


Figure 3-109

ABORT TRAJECTORY PROFILES

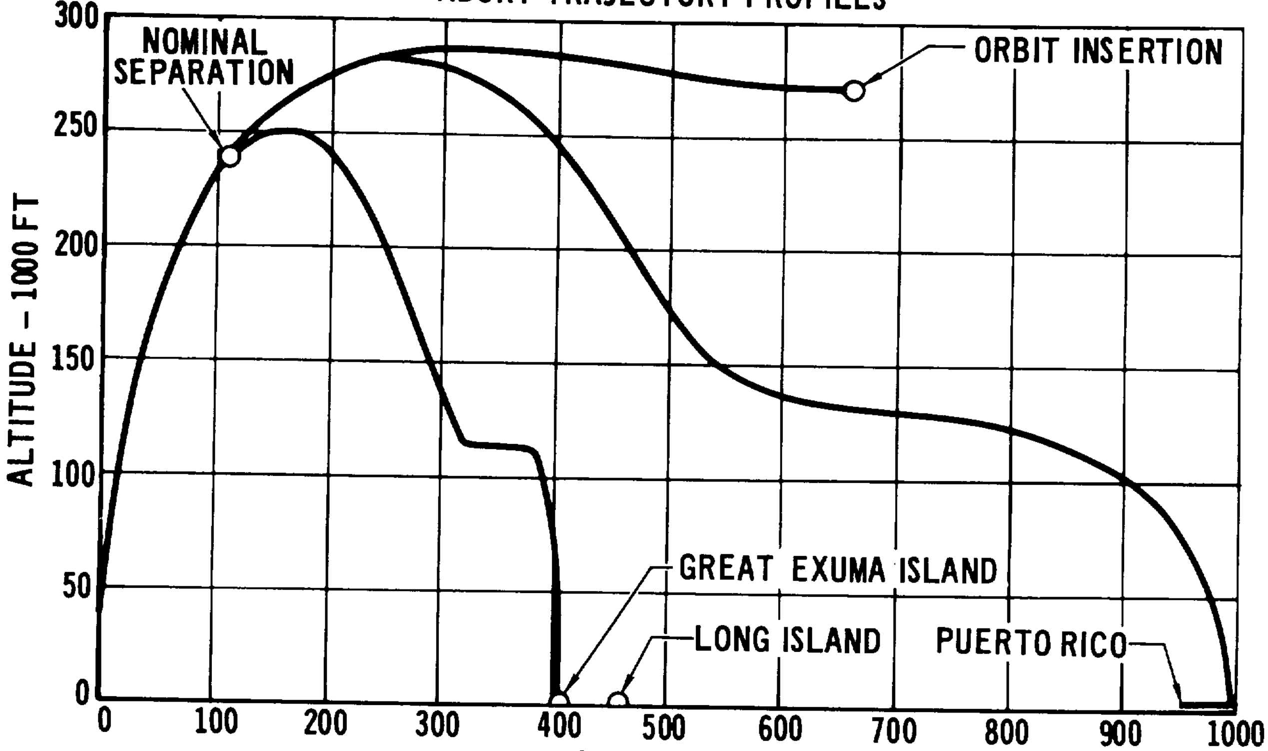


Figure 3-110

REPRESENTATIVE ABORT TRAJECTORY PARAMETERS
Part I

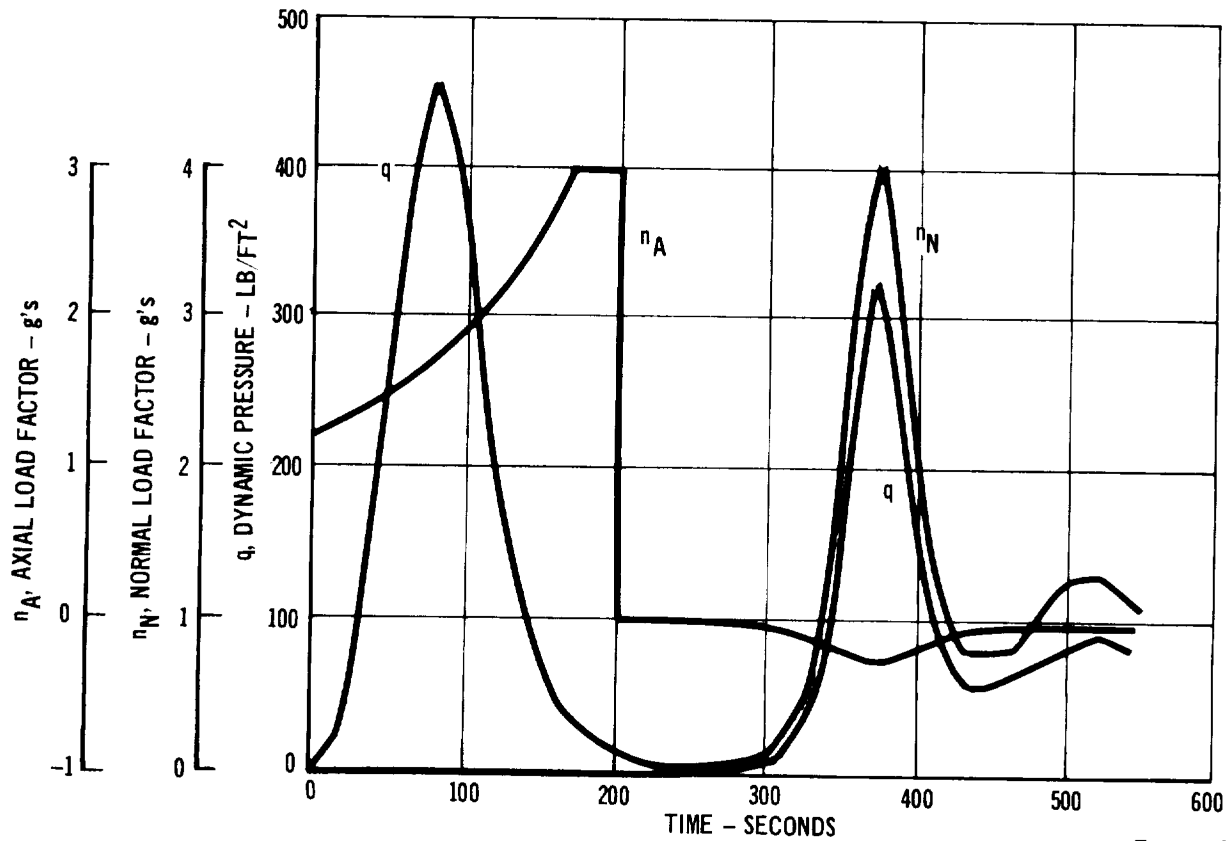


Figure 3-111

REPRESENTATIVE ABORT TRAJECTORY PARAMETERS
Part II

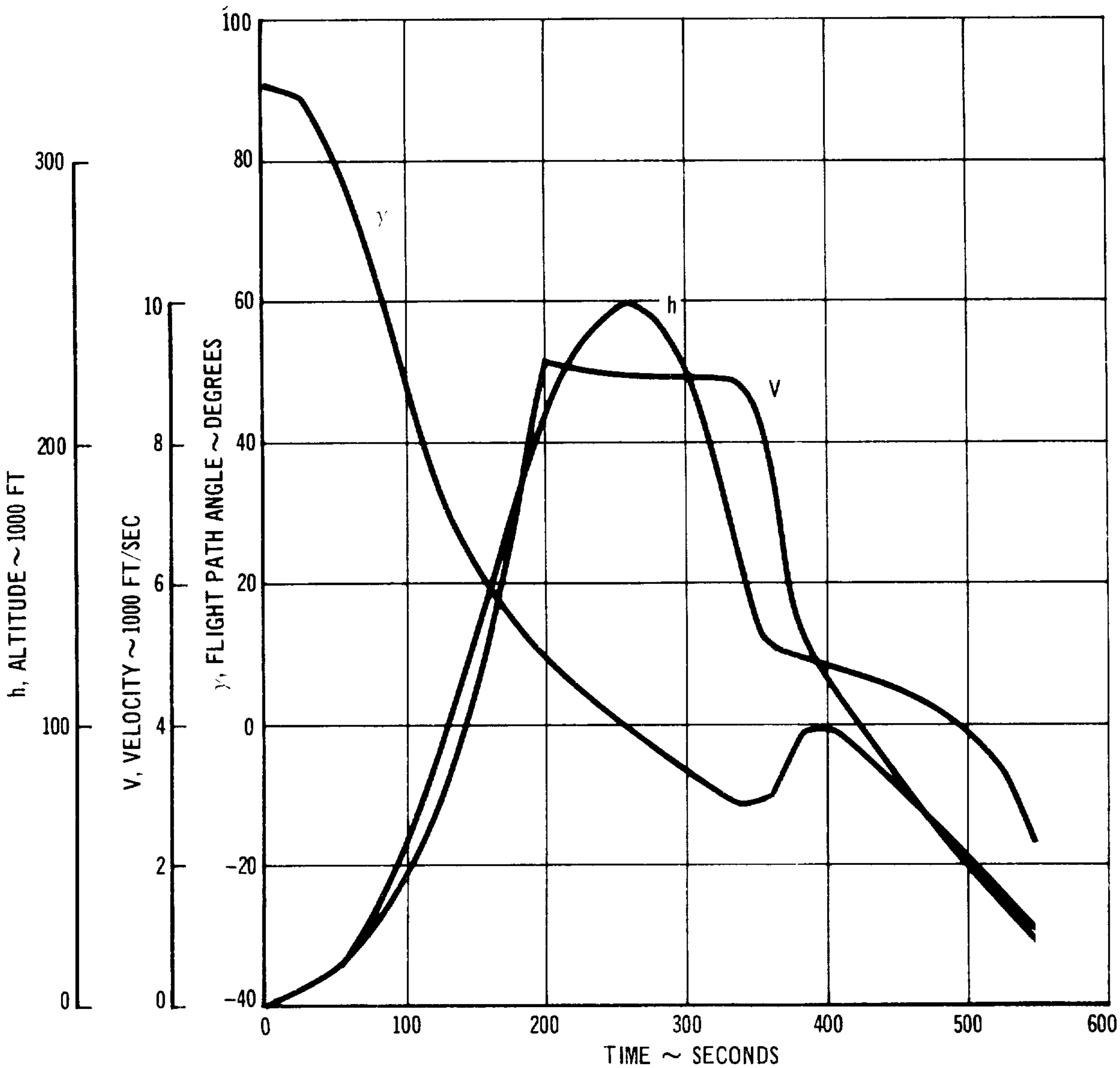


Figure 3-112

If a carrier failure occurs at any time between 20 seconds and separation the same procedure can be followed except that the weight of the HL-10 will be greater during the reentry. This will necessitate reduced angles of attack to keep the load factor at 4 g's and the temperatures will be higher. Possible range extensions may also result requiring bank angle modulation or landing at the slightly more distant Long Island instead of Great Exuma. In these cases abort to Puerto Rico as discussed below may be advantageous.

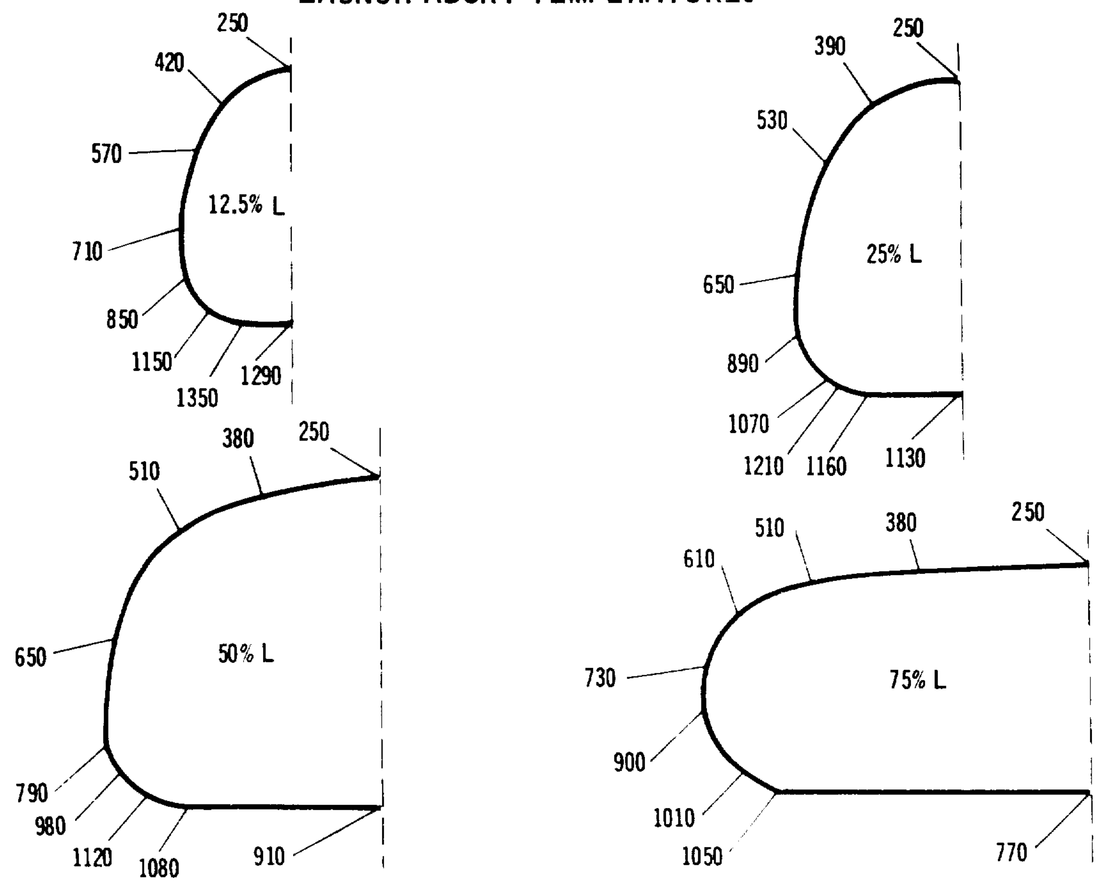
One important class of failures is Orbiter loss of power at separation or shortly thereafter. If this type of failure occurs within about 20 seconds of separation, abort to Great Exuma is still possible. For another 10 seconds after that, Long Island is attainable. If failure occurs still later neither Great Exuma nor Long Island can be reached because the flight velocities exceed that which can be dissipated by energy management.

Abort to Puerto Rico: For some failures abort to Puerto Rico is desirable and for others it is necessary. If the carrier fails in the 40 seconds prior to nominal separation it is desirable to abort to Puerto Rico because of the large fuel load which would be left on the Orbiter if the abort were to Great Exuma. In such a case the Orbiter would supply the required velocity increment. Also, if the decision to abort were made between separation and 350 seconds while the Orbiter retained some thrusting capability Puerto Rico remains a possible landing site. Another possible (but unlikely) abort to Puerto Rico would be in the case of complete Orbiter power loss between 315 and 350 seconds.

Abort to Water: Although water landing is undesirable it is worth note that the Orbiter can reach the sea in a flyable attitude if failure occurs at any time other than the first 20 seconds of flight. Trajectories have been calculated for such instances with initial velocities up to 22,000 ft/sec. Thermodynamic analysis indicates the vehicle would survive the reentry maneuver.

3.5.8 Orbiter Abort Heating Analysis - A heating analysis was conducted for the abort trajectories defined in section 3.5.7 of this volume. As stated in section 3.5.7 if an abort becomes necessary prior to staging the orbiter engines will be used to reach the normal staging point (h = 220,000 ft. and V = 9166 ft/sec). The Orbiter will then reenter to a landing at Great Exuma. The maximum surface temperatures experienced by the Orbiter during this abort reentry are presented in Figure 3-113 for four body locations. The maximum surface temperatures vary from

ORBITER
LAUNCH ABORT TEMPERATURES



NOTE:
(1) RADIATION EQUILIBRIUM TEMPERATURE ($\epsilon = 0.85$)
(2) LAMINAR HEATING
(3) NO UNCERTAINTY FACTOR
(4) $\alpha = 50^{\circ}$
(5) ABORT PRIOR TO STAGING

Figure 3-113

250°F on the upper surface to 1350°F on the lower surface.

When abort from altitudes higher than the staging altitude is necessary, more severe thermal environments result. Figure 3-114 presents maximum lower surface temperature as a function of abort velocity at 12.5% of vehicle length. The maximum lower surface temperature exceeds the 2200°F allowable for TD-Ni reuse for abort velocities above 16,500 ft/sec. However, the maximum temperatures at the reference point are, below the 2400°F maximum allowable for TD-Ni and thus only selective replacement of panels would be necessary. Temperatures presented in this section are laminar radiation equilibrium temperatures based on a surface emittance of 0.85 and the heating methods discussed in Section 3.3.2.

MAXIMUM ORBITER TEMPERATURES DURING REENTRY FROM HIGH ALTITUDE ABORT
(Lower Surface – 12.5% of Vehicle Length)

NOTE:
(1) RADIATION EQUILIBRIUM TEMPERATURES (ϵ 0.85)
(2) LAMINAR HEATING
(3) NO UNCERTAINTY FACTOR

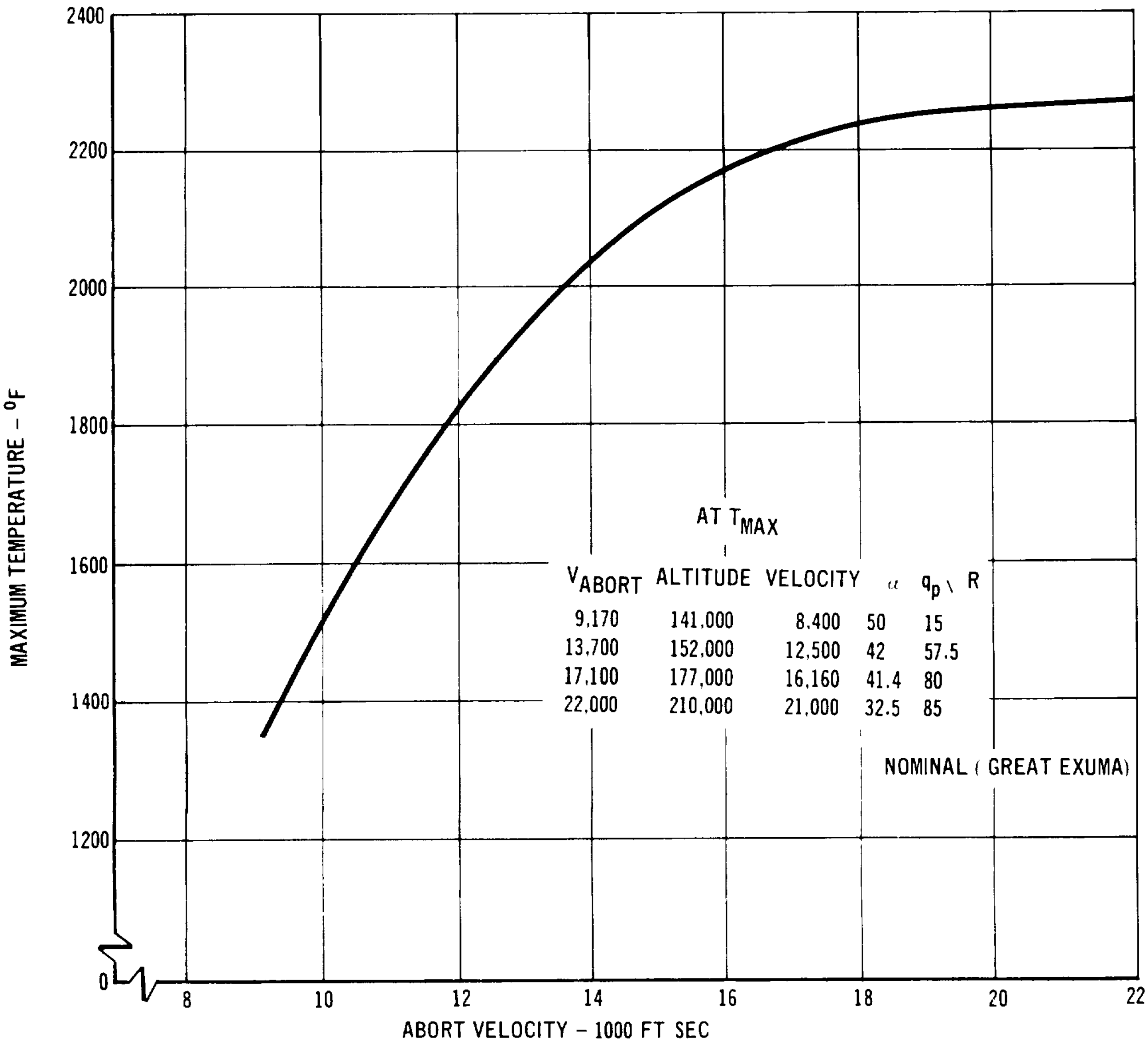


Figure 3–114

4.0 OPERATIONS ANALYSIS

The mission operational considerations including ground turnaround, mission interface, cargo handling and crew accommodations, are discussed in this section. The objective of the ground turnaround analysis was to determine the minimum required maintenance that can reasonably be achieved for the two stage reusable system. This objective was attained by preparing a detailed breakdown of the specific tasks and functions required from landing through launch and by estimating the manhours and facilities required for each tank using existing historical data. Similarly, the objective of the mission interface study was to define the major mission interfaces, identify the associated functional requirements and to evaluate alternate modes of performing the required functions. This was accomplished through a detailed enumeration of all required functions in each mission phase and selecting "best" operational modes for accomplishing these functions from the set of possible modes.

4.1 Ground Turnaround Analysis - The ground turnaround analysis is a special emphasis study and has as its objectives to identify maintenance tasks, system requirements and constraints and establish facilities, equipment and manpower requirements for the turnaround cycle. A system engineering approach is applied to achieve these objectives since it provides an orderly approach, a convenient means of documenting, and ease of understanding. The items generated in this analysis are:

- o Functional flow diagrams of the total maintenance turnaround cycle.
- o Task analysis
- o Timeline analysis

To facilitate this discussion numerous symbols and abbreviations were employed. The definitions of these symbols are summarized at the end of the section in Table 4-17.

Functional Flow - Functional flow diagrams were prepared in general form to cover all design candidate concepts that were considered. The general form was then tailored to the baseline design in three phases: (1) post flight maintenance; (2) maintenance cycle; (3) launch preparation as illustrated in Figure 4-1. These phases are discussed in detail in Sections 4.1.1, 4.1.2, and 4.1.3. In Figure 4-1 the numbers above the blocks indicate the time necessary to complete the task. The number on the left side of the block is elapsed time and the number on the right is manhours. These times were obtained through tasks analysis of the various subsystems.

FUNCTIONAL FLOW BASIC DESIGN

14.6 186	60 4042	24 856	
1.0	2.0	3.0	
Post Flight Maintenance →	Maintenance Cycle →	Launch Preparation →	Launch

Figure 4-1

Task Analysis - The Task Analysis defines the scope of each functional block regarding subtasks, the type of scheduled maintenance, frequency of the task, man-hours to complete the task, personnel required to do the job and the elapsed time. Where necessary, the functional flows are broken down to the fifth and sixth level identifying such components as values, tubing, wire bundles, engine nozzles, etc.

Time Line Analysis - Figure 4-2 illustrates the minimum turnaround summary time line analysis for both the carrier and the orbiter. The results of the functional flows and task analyses indicate that it will take 360 men 17,076 man-hours to complete the ground turnaround cycle in six (6) days. During postflight and the maintenance cycle phases, two (2) 8-hour shifts will be worked per day. Three (3) 8-hour shifts of continuous operation will be necessary for launch preparation. Figure 4-3 depicts the various elements of the turnaround cycle. Table 4-6 provides a breakdown of manhour utilization.

The Maintenance Control organization, discussed in the maintenance plan provides efficient utilization of the work force by planning, scheduling, and controlling all spacecraft maintenance so that peak maintenance periods are staggered and high quality maintenance can be performed at all times. Facility and equipment requirements for maintenance support are shown in Table 4-1.

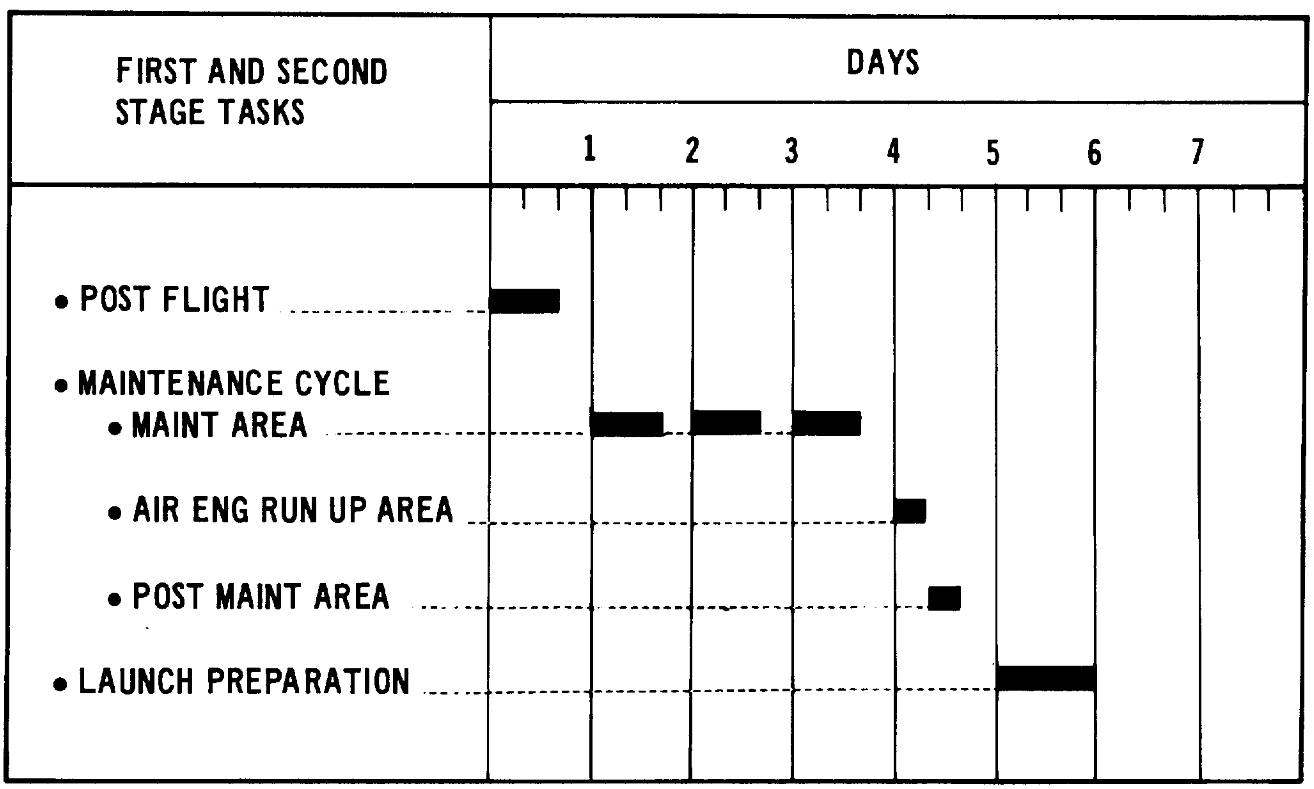
4.1.1 Post Flight Maintenance - A detailed analysis was conducted to identify necessary maintenance tasks performed during the post flight phase. Results of the Post Flight Phase Analysis are indicated in Table 4-2. A remote area will be provided for deservicing. This area will be equipped with an overhead monorail to remove the payload. Post flight maintenance consists of 30 maintenance tasks requiring 14.6 elapsed hours.

The post flight functional flow diagrams are illustrated in Figure 4-4. A brief description of each task is outlined in the following paragraphs. The number to the right of the title corresponds to the functional flow block. The post flight timeline is shown in Figure 4-5.

Post Flight Maintenance (1.0) - See Figure 4-4.

Crew Egress and Data Removal (1.1) - This task commences immdeidately after the vehicle is parked outside of the service area, the assist engines are shut down and the crew egress stands are positioned at the hatch. As the crew leaves the vehicle they will remove the onboard checkout tapes, flight recorder tapes and the flight-log.

MINIMUM TURNAROUND SUMMARY



ILRVS-389F

Figure 4-2

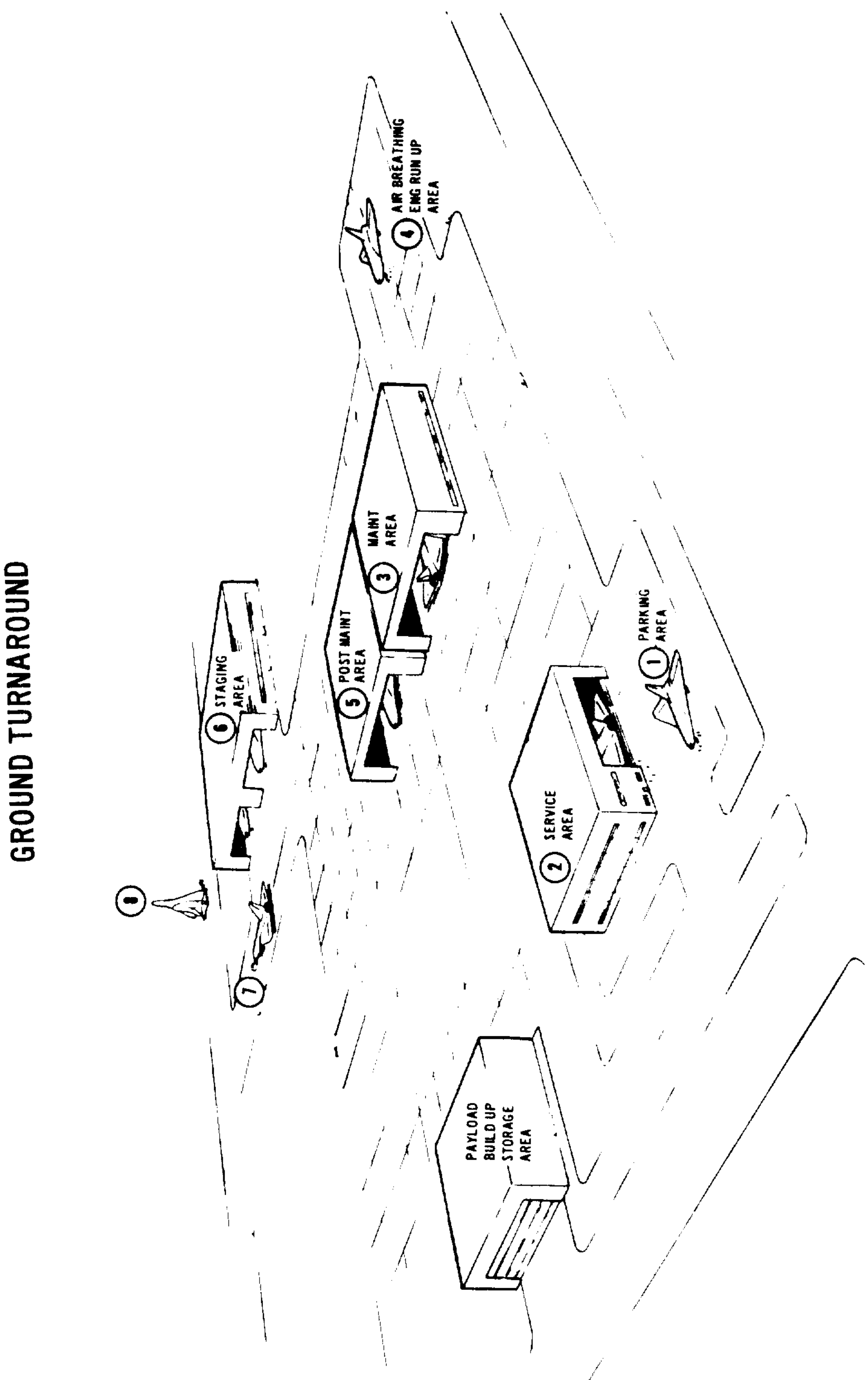

GROUND TURNAROUND
1 PARKING AREA
2 SERVICE AREA
3 MAINT AREA
4 AIR BREATHING ENG RUN UP AREA
5 POST MAINT AREA
6 STAGING AREA
7
8
PAYLOAD BUILD UP STORAGE AREA

Figure 4–3

Table 4-1

MAINTENANCE FACILITY REQUIREMENTS

AREA	FACILITY REQUIREMENT	EQUIPMENT REQUIREMENT
PARKING	• SPACE AND LOAD BEARING CAPABILITY OF SUPPORT FOR FIRST AND SECOND STAGE VEHICLES	• CREW EGRESS VEHICLE • JP FUEL TRUCK
SERVICE	• PAYLOAD REMOVAL CAPABILITY • OXYGEN AND HYDROGEN FUEL DESERVICING CAPABILITY	• PRIME MOVER VEHICLE • PAYLOAD TRAILER
MAINTENANCE	• VEHICLE ACCESS EQUIPMENT • ELECTRICAL POWER • HYDRAULIC POWER • PNEUMATIC SERVICE • EC/LSS SERVICE • SANITATION SERVICE • PURGE EQUIPMENT • SAFETY EQUIPMENT • ELECTRICAL POWER SHOP • COMM/NAV EQUIPMENT SHOP • GUIDANCE & CONTROL EQUIPMENT SHOP • EC/LSS EQUIPMENT SHOP • HYDRAULIC EQUIPMENT SHOP • ASSIST ENGINES SHOP • MAIN PROPULSION AND MANEUVERING ENGINES SHOP	• ENGINE DOLLYS AND STANDS • MAIN ENGINE REMOVAL AND INSTALLATION VEHICLE • MAIN ENGINE STANDS
ASSIST ENGINES RUNUP	• JET ENGINE RUNUP PAD	• J P FUEL TRUCK
POST MAINTENANCE	• VEHICLE ACCESS EQUIPMENT • ELECTRICAL POWER • HYDRAULIC POWER • PNEUMATIC SERVICE • EC/LSS SERVICE • SAFETY EQUIPMENT	• OVERHEAD MONORAIL
STAGING AREA	• PROTECTED ENVIRONMENT FOR SPACECRAFT STORAGE	
PAYLOAD BUILD UP		
VEHICLE MATING AREA	• HIGH BAY AREA (VAB) • TRANSPORTER (LUT)	• OVERHEAD CRANE • CHECKOUT EQUIPMENT
PAD	• ERECTION EQUIPMENT • CRYOGENIC SERVICING	• FINAL CHECKOUT EQUIPMENT

Table 4-2

WHAT WAS LEARNED FROM POST FLIGHT ANALYSIS

- An area is required for spacecraft cooling.

  Reason - Spacecraft surfaces are too hot to touch immediately after landing.

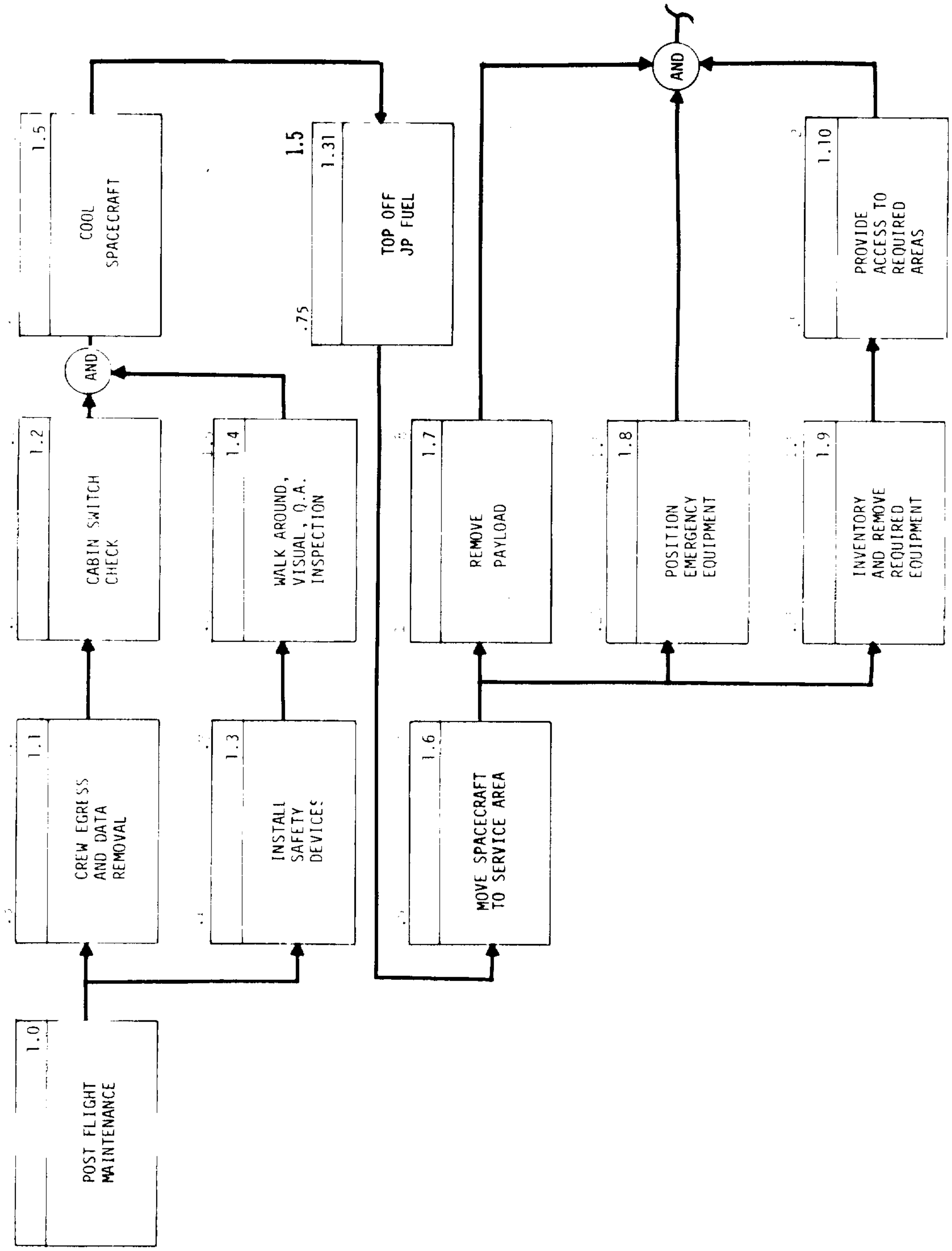
- A special area is provided for deservicing.

  Reason - Hydrogen vented into the air is a fire hazard. Special plumbing will be available to carry the down loaded hydrogen away from the service area for burning.

- Post flight can be accomplished within 14.6 hours, consuming 186 direct manhours.

  Reason - Only necessary maintenance is performed and a centralized Maintenance Control organization schedules tasks and personnel on a non-interfering basis.

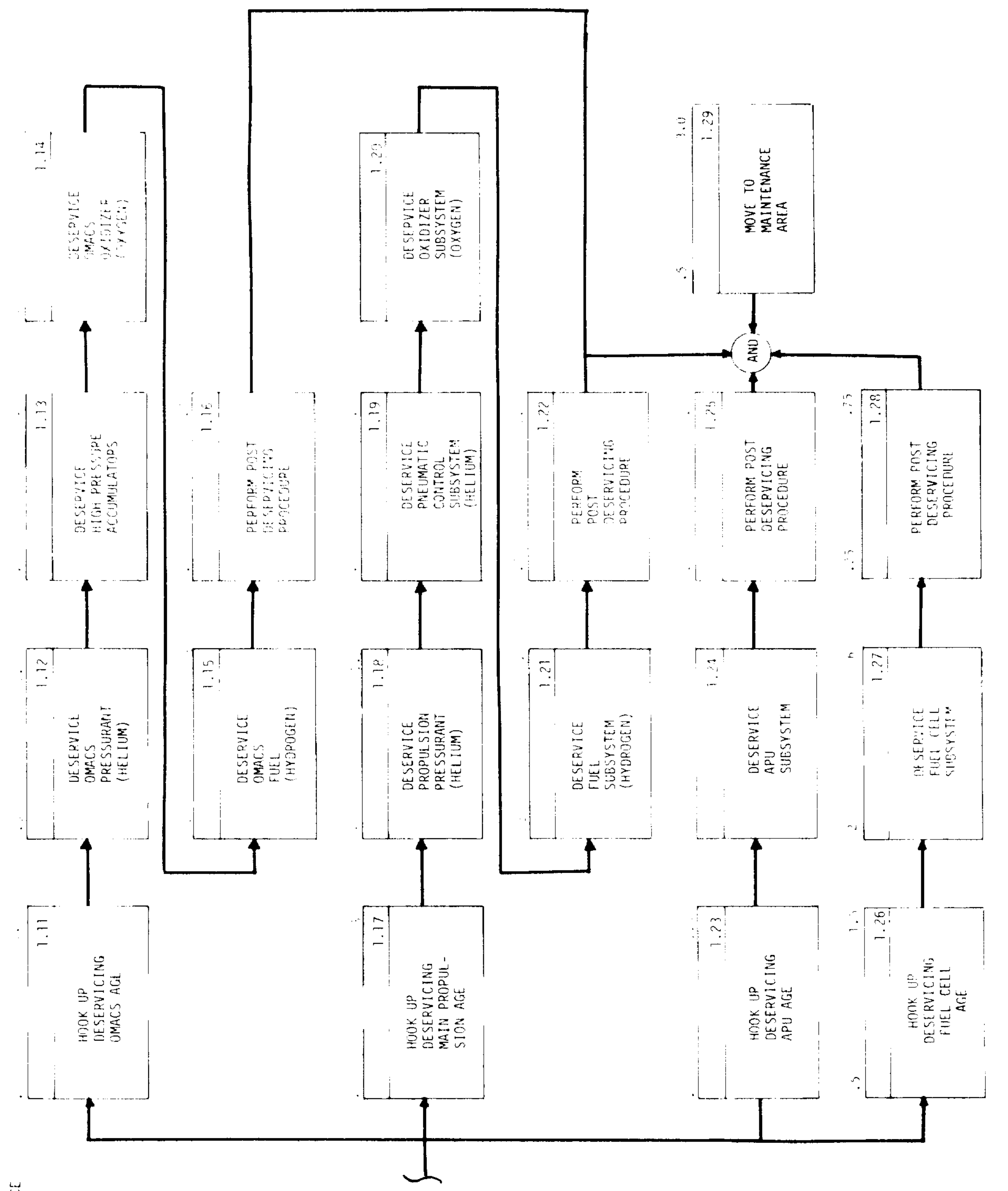
POST FLIGHT MAINTENANCE



POST FLIGHT MAINTENANCE

Sheet 1 of 9
Figure 4-4

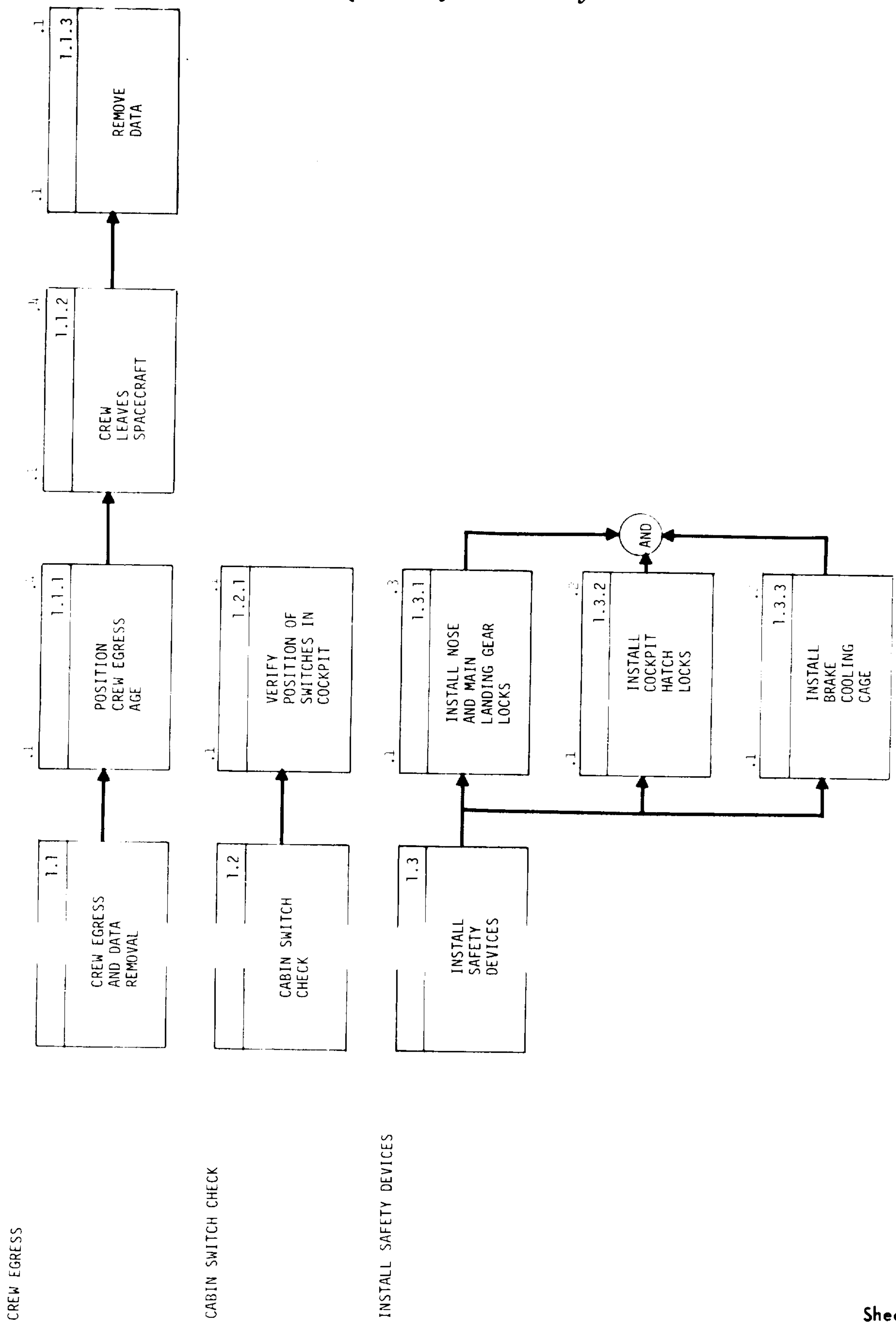
POST FLIGHT MAINTENANCE (CONTINUED)



POST FLIGHT MAINTENANCE

Sheet 2 of 9
Figure 4-4

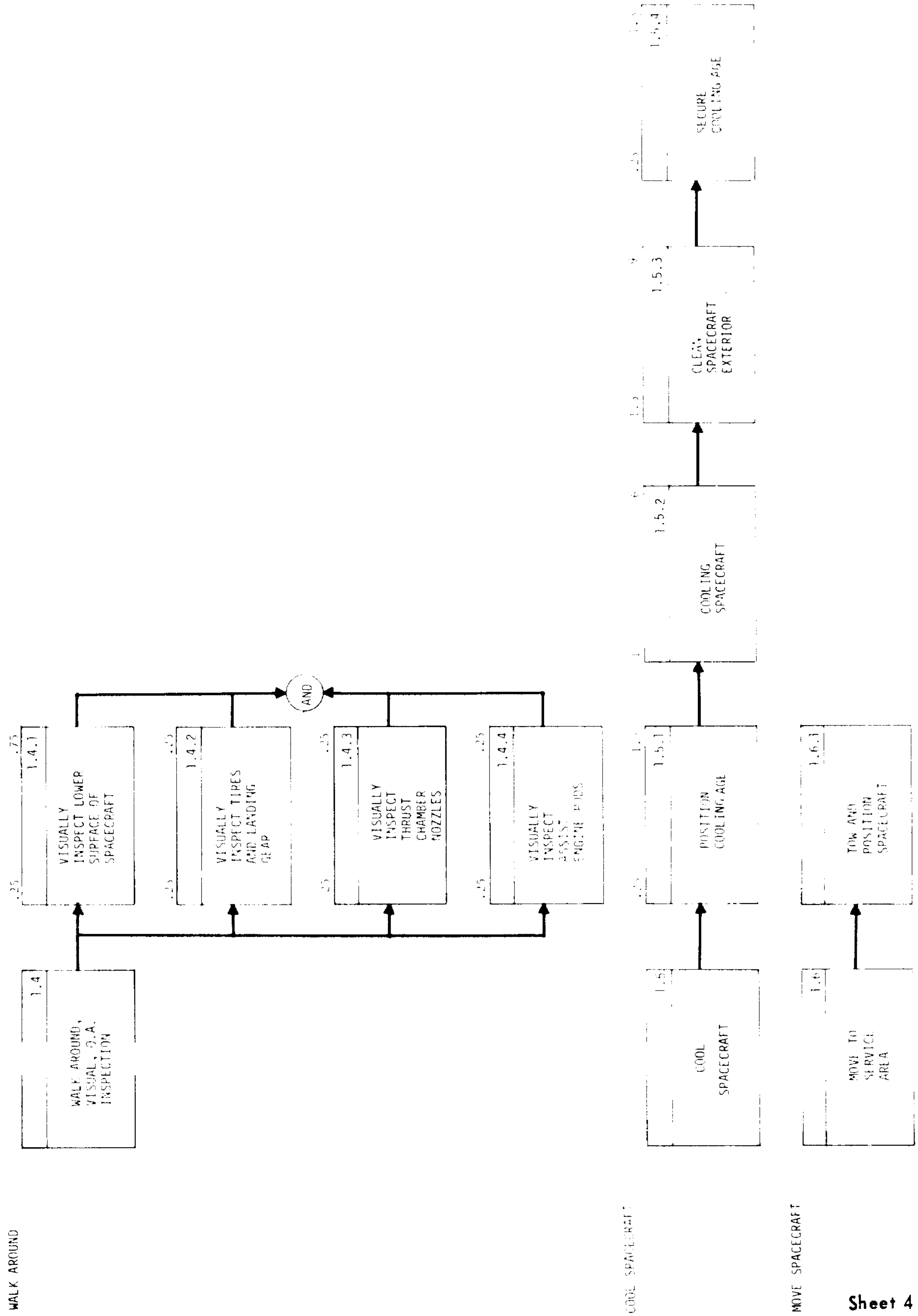
POAT FLIGHT MAINTENANCE (CONTINUED)



Sheet 3 of 9

Figure 4-4

POST FLIGHT MAINTENANCE (CONTINUED)



Sheet 4 of 9

Figure 4-4

POST FLIGHT MAINTENANCE (CONTINUED)

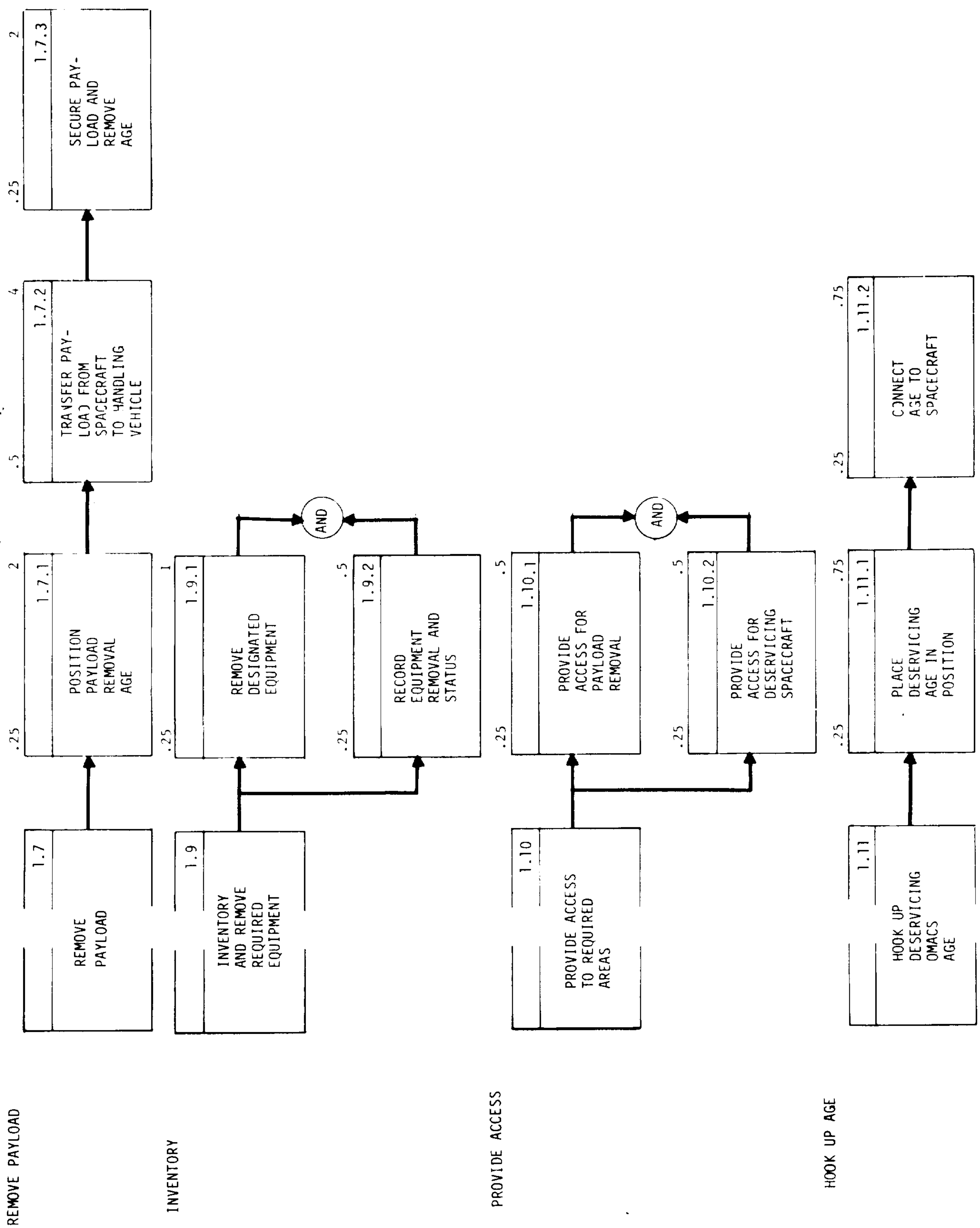


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Figure 4–4

POST FLIGHT MAINTENANCE (CONTINUED)

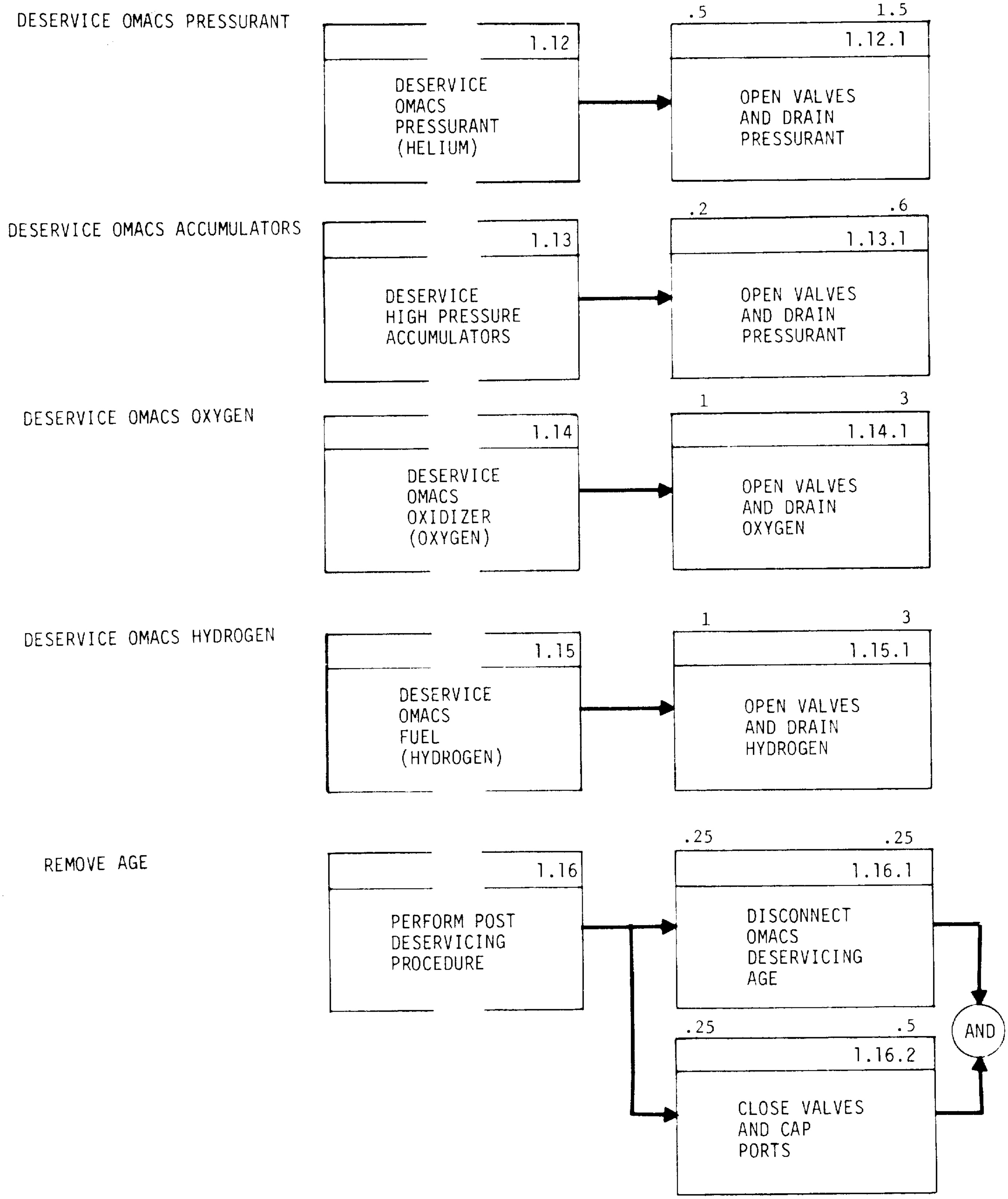

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Figure 4-4

POST FLIGHT MAINTENANCE (CONTINUED)

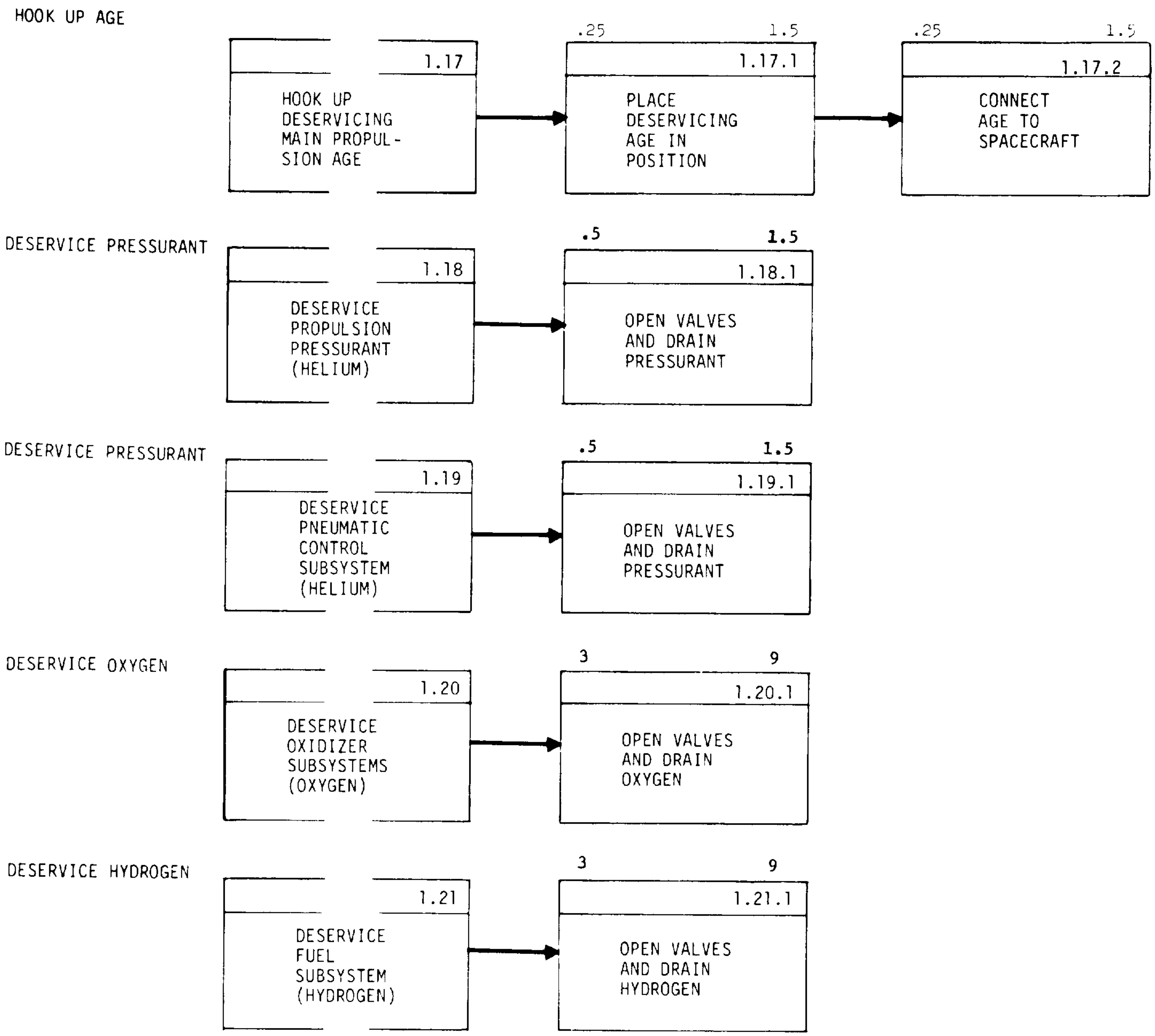

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Figure 4-4

POST FLIGHT MAINTENANCE (CONTINUED)

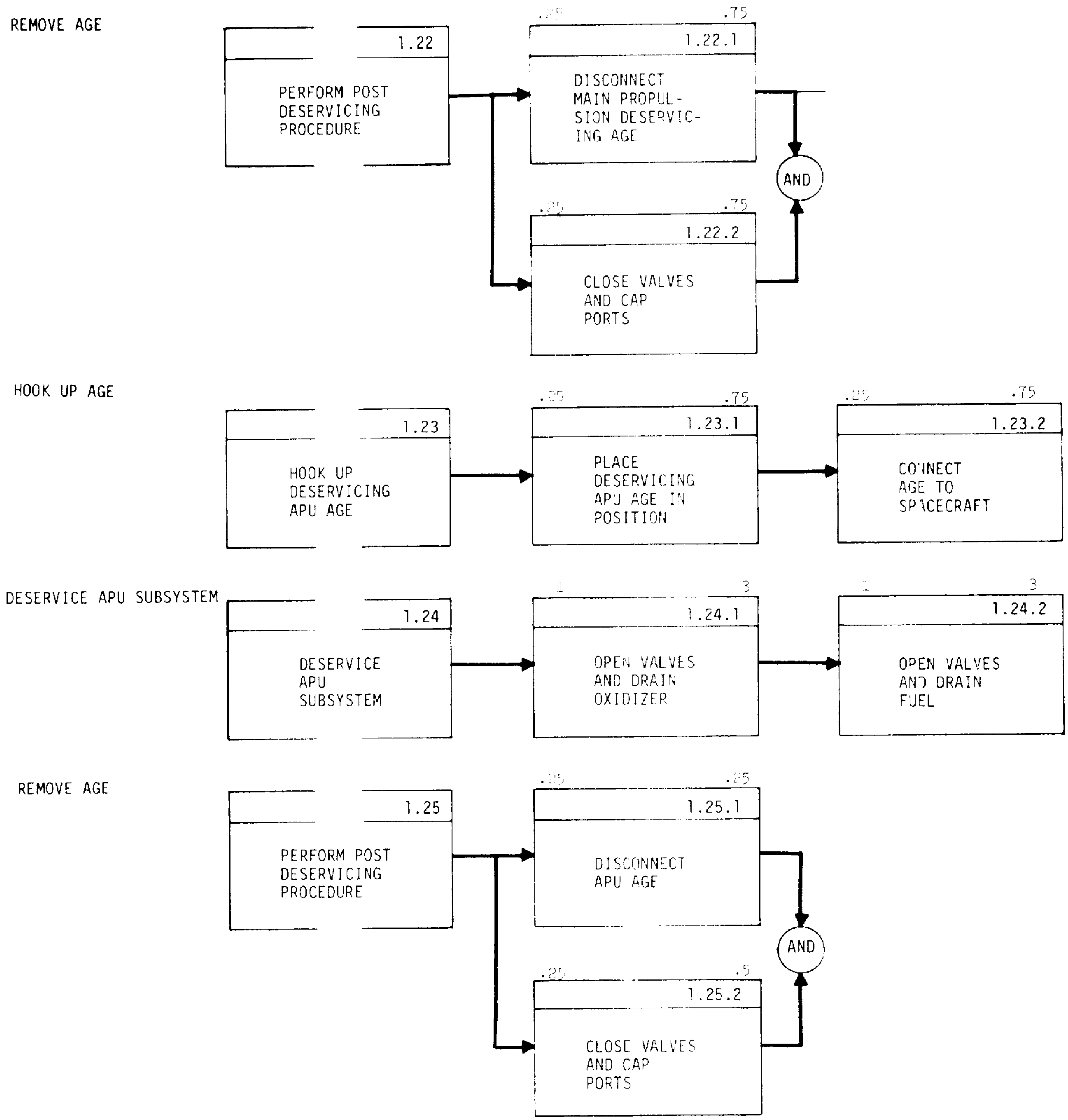

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Figure 4-4

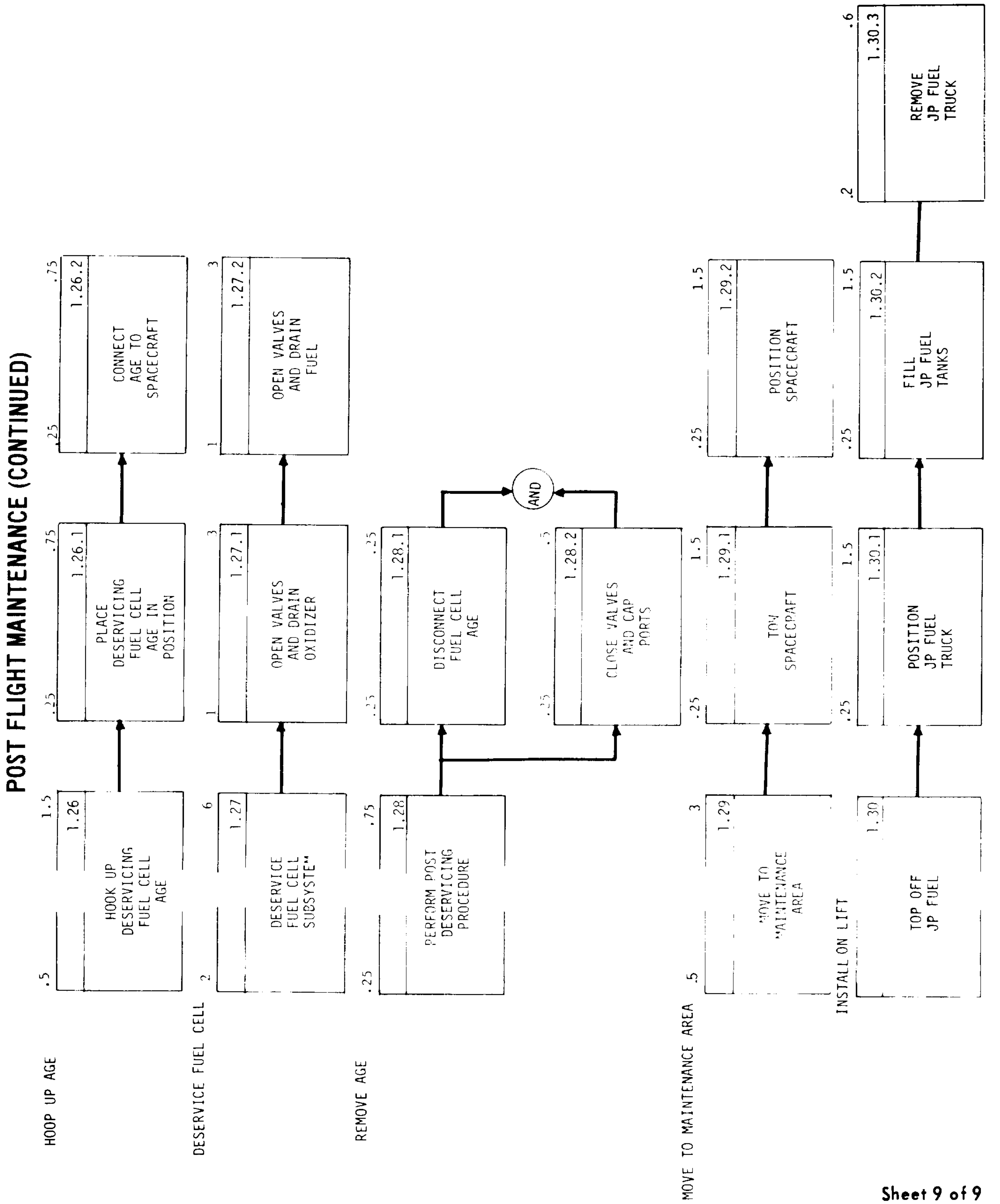

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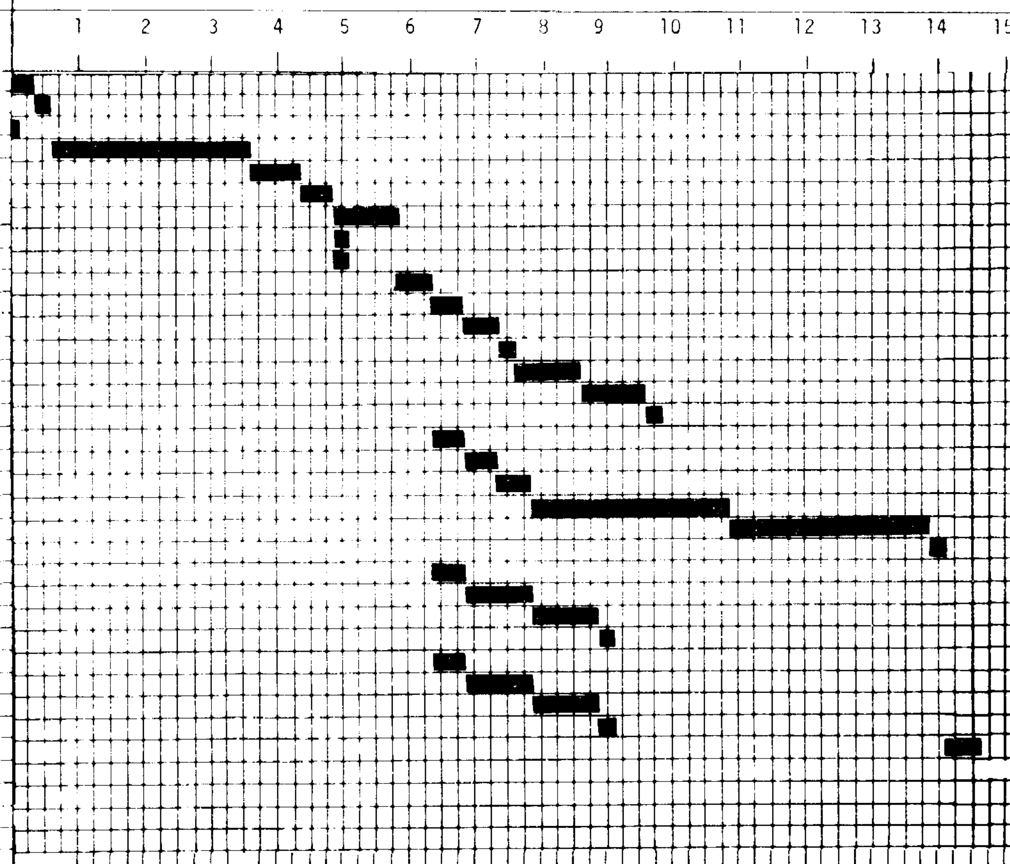
Figure 4-4

POST FLIGHT TIMELINE ANALYSIS

PHASE	LOCATION	TYPE OF MAINTENANCE
POST FLIGHT MAINTENANCE	POST FLIGHT AREA	COOL DOWN, PAYLOAD REMOVAL, DESERVICING

ELAPSED TIME - HOURS



FUNCTION	FUNCTION AND CORRESPONDING TASKS	M/H
1	CREW EGRESS AND DATA REMOVAL	.9
2	CABIN SWITCH CHECK (QA INSPECTION)	1.6
3	INSTALL SAFETY DEVICES	.7
4	COOL DOWN SPACECRAFT	18.0
5	TOP OFF JP FUEL	
6	MOVE TO SERVICE AREA	3.0
7	REMOVE PAYLOAD	8.0
8	POSITION EMERGENCY EQUIPMENT	1.5
9	INVENTORY AND REMOVE REQUIRED EQUIPMENT	1.5
10	PROVIDE ACCESS TO REQUIRED AREAS	1.
11	HOOK UP DESERVICING OMACS AGE	1.5
12	DESERVICE OMACS PRESSURANT	1.5
13	DESERVICE HIGH PRESSURE ACCUMULATORS	.6
14	DESERVICE OMACS OXIDIZER	3.0
15	DESERVICE OMACS FUEL	3.0
16	PERFORM POST DESERVICING TASK	.75
17	HOOK UP DESERVICING MAIN PROPULSION AGE	3.0
18	DESERVICE MAIN PRESSURANT	1.5
19	DESERVICE PHENUMATIC CONTROL SUBSYSTEM	1.5
20	DESERVICE MAIN OXIDIZER SYSTEM	9.0
21	DESERVICE MAIN FUEL SYSTEM	9.0
22	PERFORM POST DESERVICE TASK	1.5
23	HOOK UP APU AGE	1.5
24	DESERVICE OXIDIZER SYSTEM (APU)	3.0
25	DESERVICE FUEL SYSTEM (APU)	3.0
26	PERFORM POST DESERVICE TASKS	.75
27	HOOK UP FUEL CELL AGE	1.5
28	DESERVICE FUEL CELL OXIDIZER	3.0
29	DESERVICE FUEL CELL	3.0
30	PERFORM POST DESERVICING TASKS	.7
31	MOVE SPACECRAFT TO MAINTENANCE AREA	3.0

Figure 4-5

Cabin Switch Check (1.2) - The position of each switch is documented and delivered to the maintenance debriefing section as an aid in comprehensively diagnosing possible malfunctions. The Maintenance debriefing section is discussed in detail in the Maintenance Plan.

Install Safety Devices (1.3) - Prior to departure from the crew department, a crew member will depress the automatic safety locking switch. This switch safes all critical items which, if failed while unsafe, could cause equipment damage or injury to personnel. Mechanical devices must be installed before routine maintenance can be performed on the vehicle.

Walk Around Visual QA Inspection (1.4) - Quality assurance will perform an immediate visual inspection of the outside surfaces, engines, landing gears, etc. Any unusual discrepancies will be reported to Maintenance Control. Normal or minor discrepancies discovered at this time will be submitted to maintenance debriefing.

Cool Spacecraft (1.5) - Since the spacecraft absorbs heat while entering the earth's atmosphere, it will require three hours to cool before maintenance can be performed. Cooling Aerospace Ground Equipment (AGE) can be used to accelerate the cool down time.

Top Off JP Fuel (1.30) - The JP fuel tank will be filled to capacity prior to engering the service area in order to comply with safety directives.

Move Spacecraft to Service Area (1.6) - A six man crew is required to move the spacecraft into the service area. Following removal of the wheel chocks, one man will ride the spacecraft's brakes, one man on each wing, one man at the nose and another at the tail and one will operate the tow tug. Once the spacecraft is in the service area, the brakes will be set, the wheel chocks positioned, the fire extinguishers positioned at the designated areas, and all mechanical devices (e.g., clam shells, brace bar locks, etc.) installed.

Remove Payload (1.7) - A trailer with payload container cradles will be positioned adjacent to the spacecraft. With a power cable plugged into the spacecraft, the payload doors will be opened by depressing the payload door switch. An eight-man crew is required to perform the payload removal task. The payload is lifted by the overhead monorail hoist and installed on the trailer cradle.

Position Emergency Equipment (1.8) - All emergency equipment necessary will be available in the service area and positioned prior to deservicing tasks.

Inventory and Remove Equipment (1.9) - All equipment used in support of the mission and not a part of the payload (e.g., cameras, recorded data, etc.) will be inventoried, removed and processed to interested activities.

Provide Access to Required Areas (1.10) - A four-man crew will provide access to the deservicing valves and vents by removing required panels.

Deservicing Orbital Maneuver ACS (1.11 - 1.16) - Once access is provided and OMACS AGE positioned, the OMACS deservicing crew will hook up the AGE and proceed to deservice the helium pressurant which will be dumped into a storage container for reuse. The high pressure accumulator oxygen valves will be opened to vent the oxygen into the atmosphere while the accumulated hydrogen is pumped into a remote area for burning. Upon completion of OMACS deservicing, the AGE lines are disconnected, all valves are closed and all ports installed.

Deservice Main Propulsion System (1.17 - 1.22) - The procedure for deservicing of OMACS also applies to the main propulsion system except that there are no high pressure accumulators. The main propulsion system, however, requires a pneumatic control system operated with helium. The helium will be dumped into storage tanks for reuse. Upon completion of this task, all valves are closed and cap ports installed.

Deservice Auxiliary Power Equipment (1.26 - 1.28) - The same procedure for APU deservicing can be used for fuel cell deservicing after fuel cell AGE is positioned.

Move Spacecraft to Maintenance Area (1.29) - With the completion of all deservicing the spacecraft is towed to the maintenance area and positioned. The brakes will be set and the fire extinguishers placed in designated areas. Prior to any maintenance, an inspection of mechanical safety devices will be conducted by Quality Assurance complying with all safety devices.

4.1.2 Maintenance Cycle - Upon completion of the post flight phase and with the vehicle positioned in the Maintenance Area, the detail analysis indicates that the maintenance performed during this phase can be completed within approximately 60 elapsed hours. Results of the Maintenance Cycle phase analysis are indicated in Table 4-3.

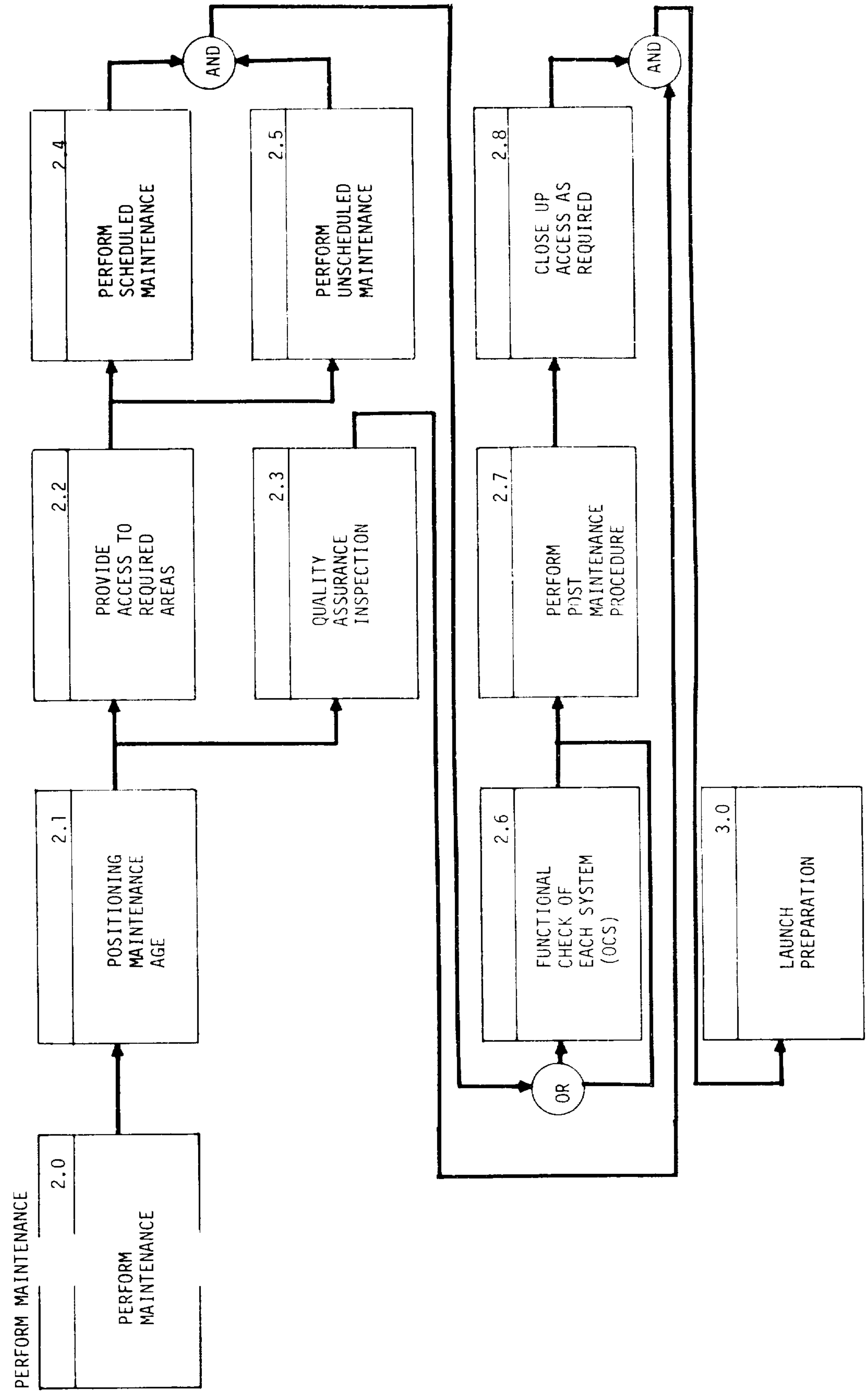
The functional flow diagrams of tasks completed in the Maintenance Area are illustrated in Figure 4-6.

Table 4-3

WHAT WAS LEARNED FROM MAINTENANCE CYCLE ANALYSIS

- A maintenance area is required.
- Reason - To provide the proper environment for maintenance personnel to perform scheduled and unscheduled maintenance, detailed visual inspection, functional checks, etc.
- An assist engine run if one is required.
- Reason - To safely run up all or any of the assist engines following completion of work conducted in the maintenance area.
- Post maintenance area is required.
- Reason - An area shall be provided to perform tasks such as install payload, service spacecraft (except cryogenics), install vehicle on erection dolly.
- The maintenance cycle can be accomplished within 60 hours consuming 4042 direct manhours, for both the Carrier and Orbiter.
- Reason - Only necessary maintenance is performed and a centralized Maintenance Control organization schedules tasks and personnel on a non-interfering basis.

MAINTENANCE CYCLE



NOTE:
1 ONBOARD CHECKOUT SYSTEM (OCS)

Sheet 1 of 31

Figure 4-6

MAINTENANCE CYCLE (Continued)

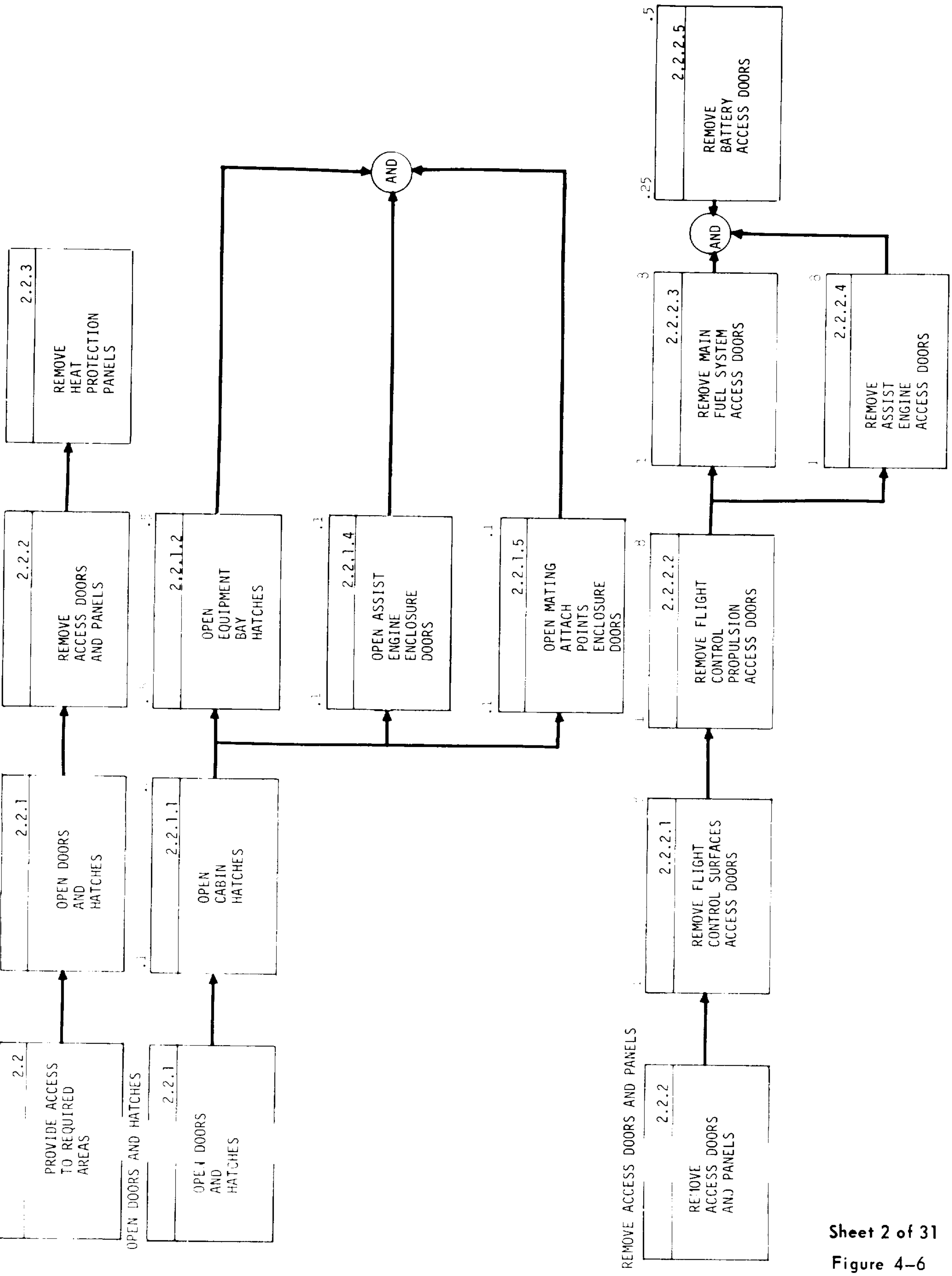


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Figure 4-6

MAINTENANCE CYCLE (Continued)

REMOVE HEAT PROTECTION PANELS

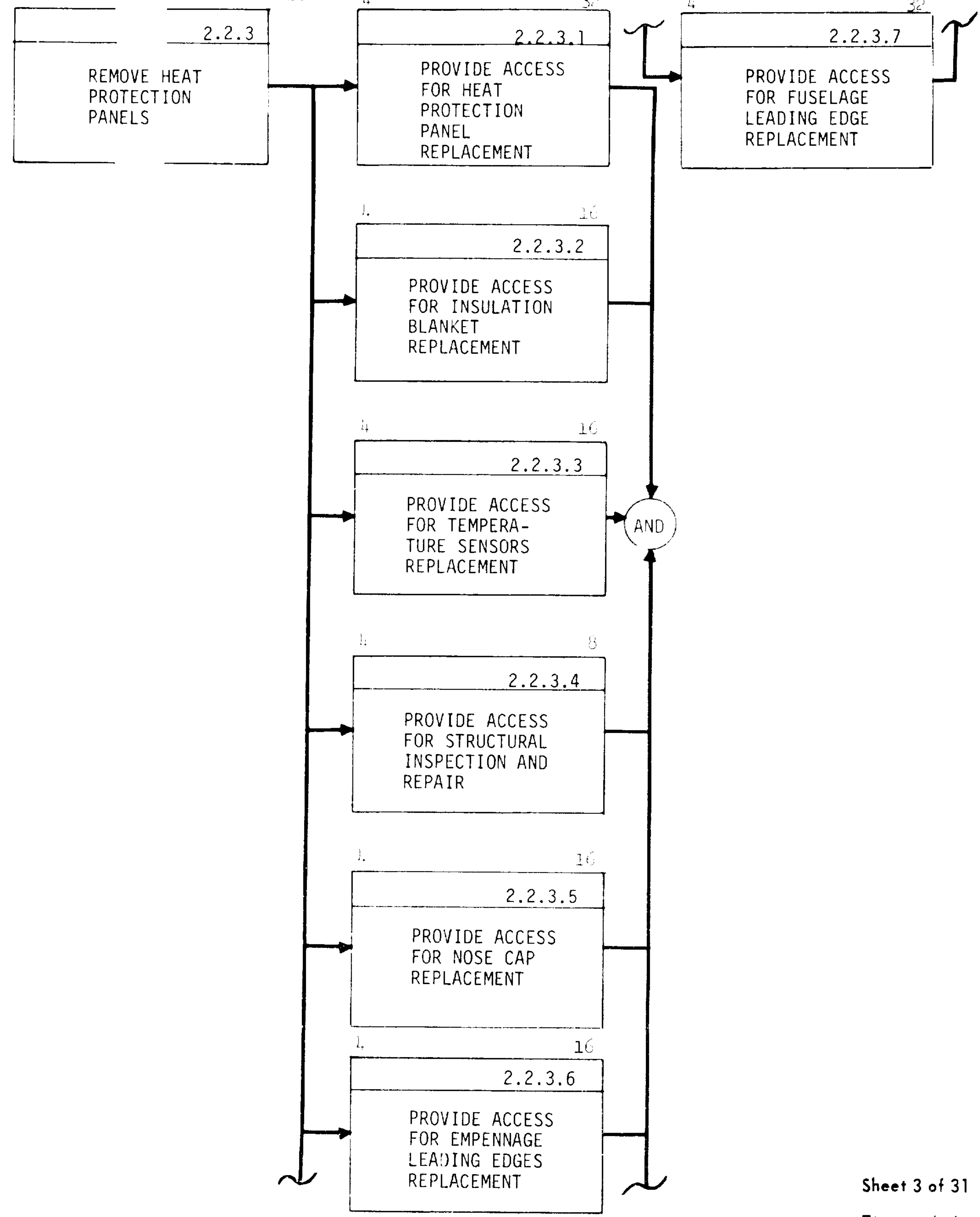

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Figure 4-6

MAINTENANCE CYCLE (Continued)

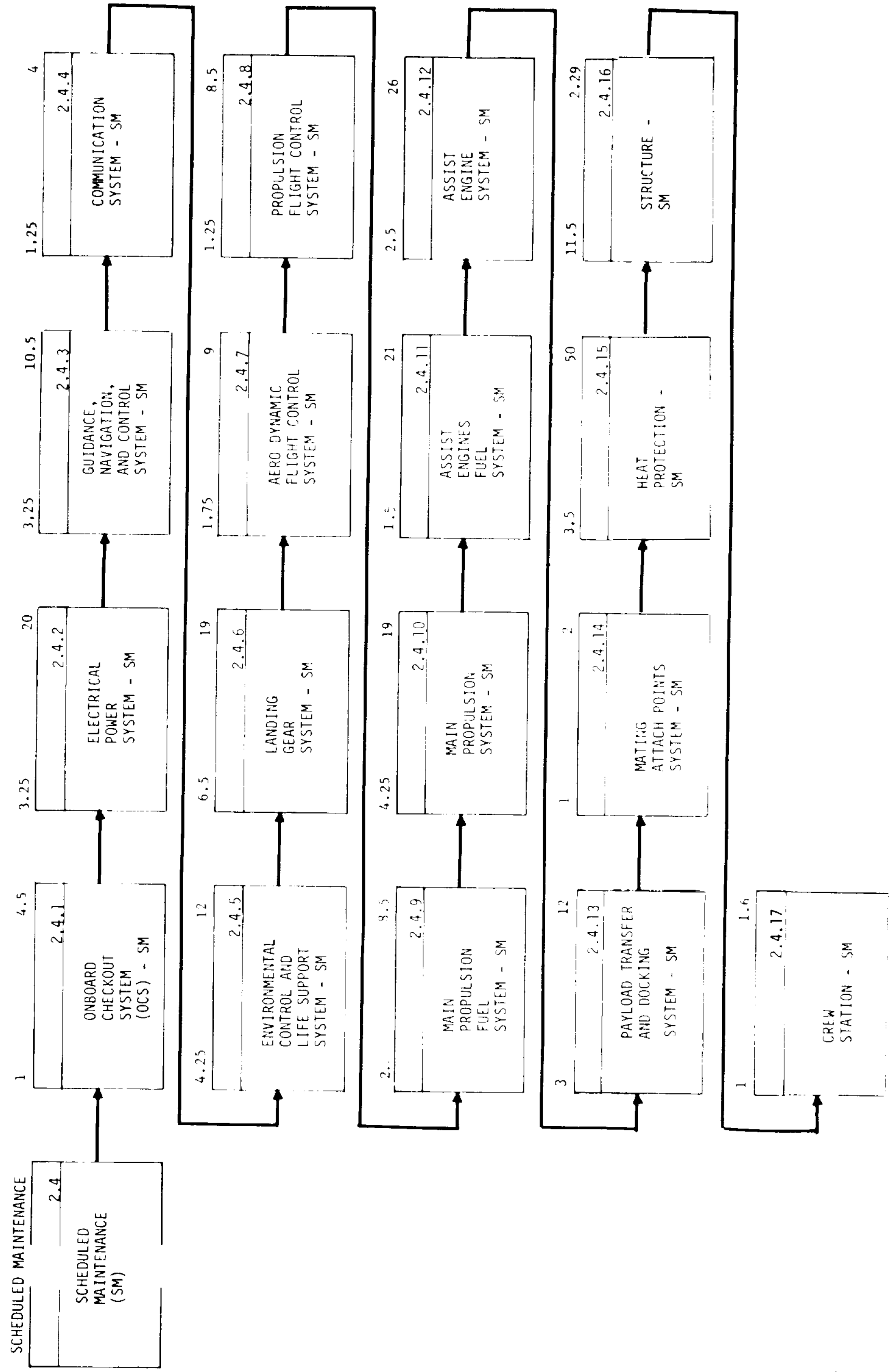

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Figure 4-6

MAINTENANCE CYCLE (Continued)

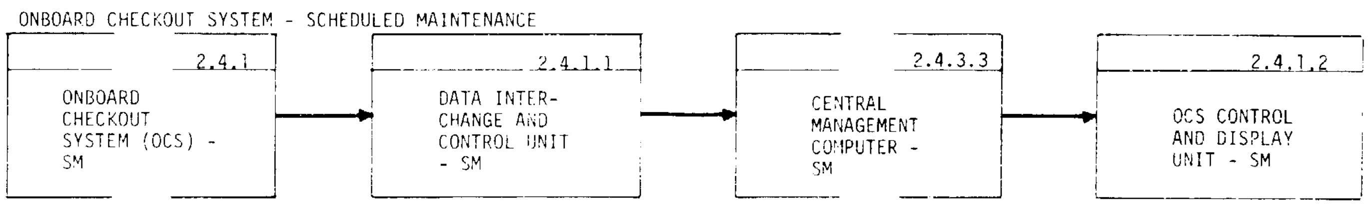

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Figure 4-6

MAINTENANCE CYCLE (Continued)

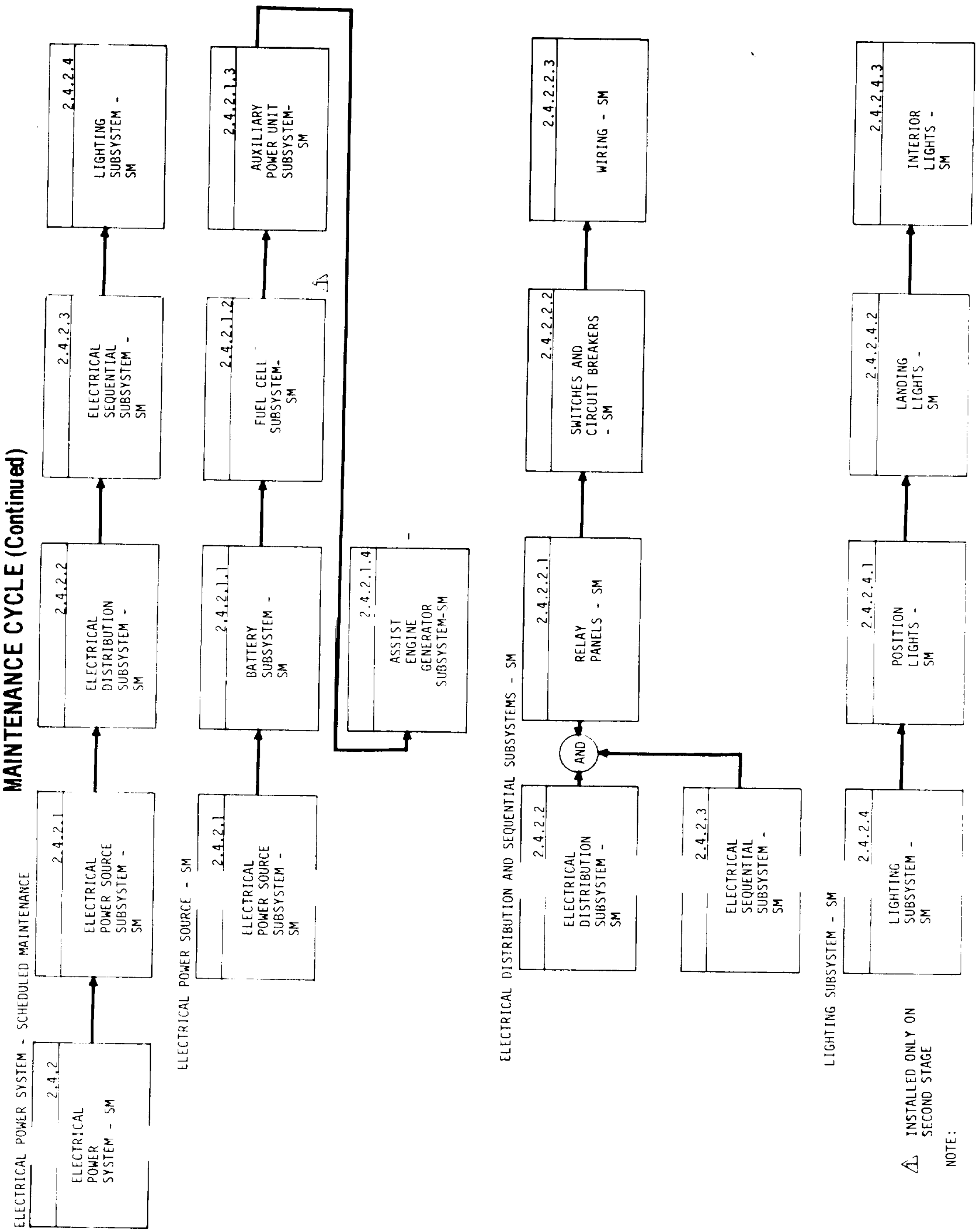

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Figure 4-6

MAINTENANCE CYCLE (Continued)

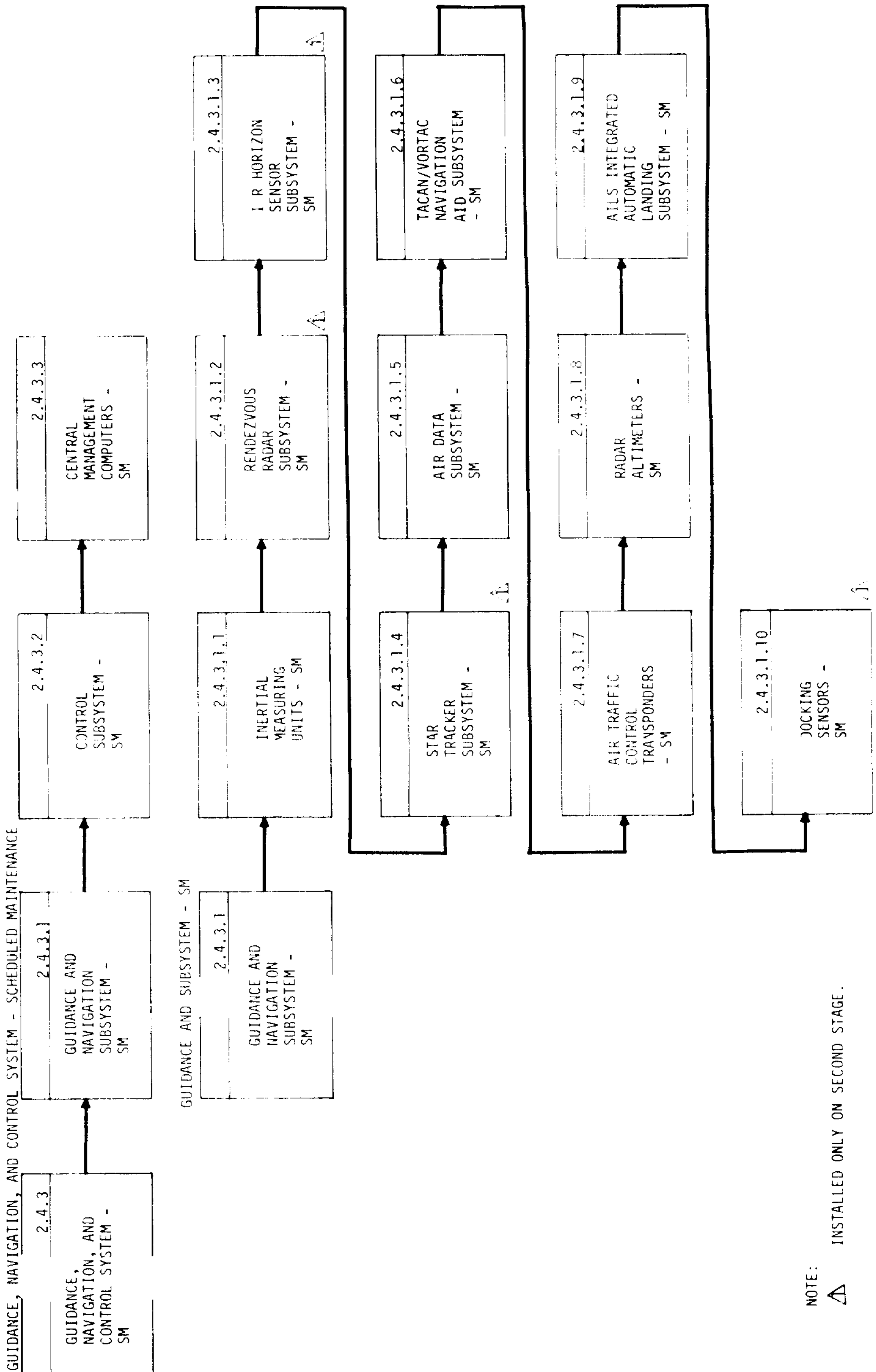

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Figure 4-6

MAINTENANCE CYCLE (Continued)

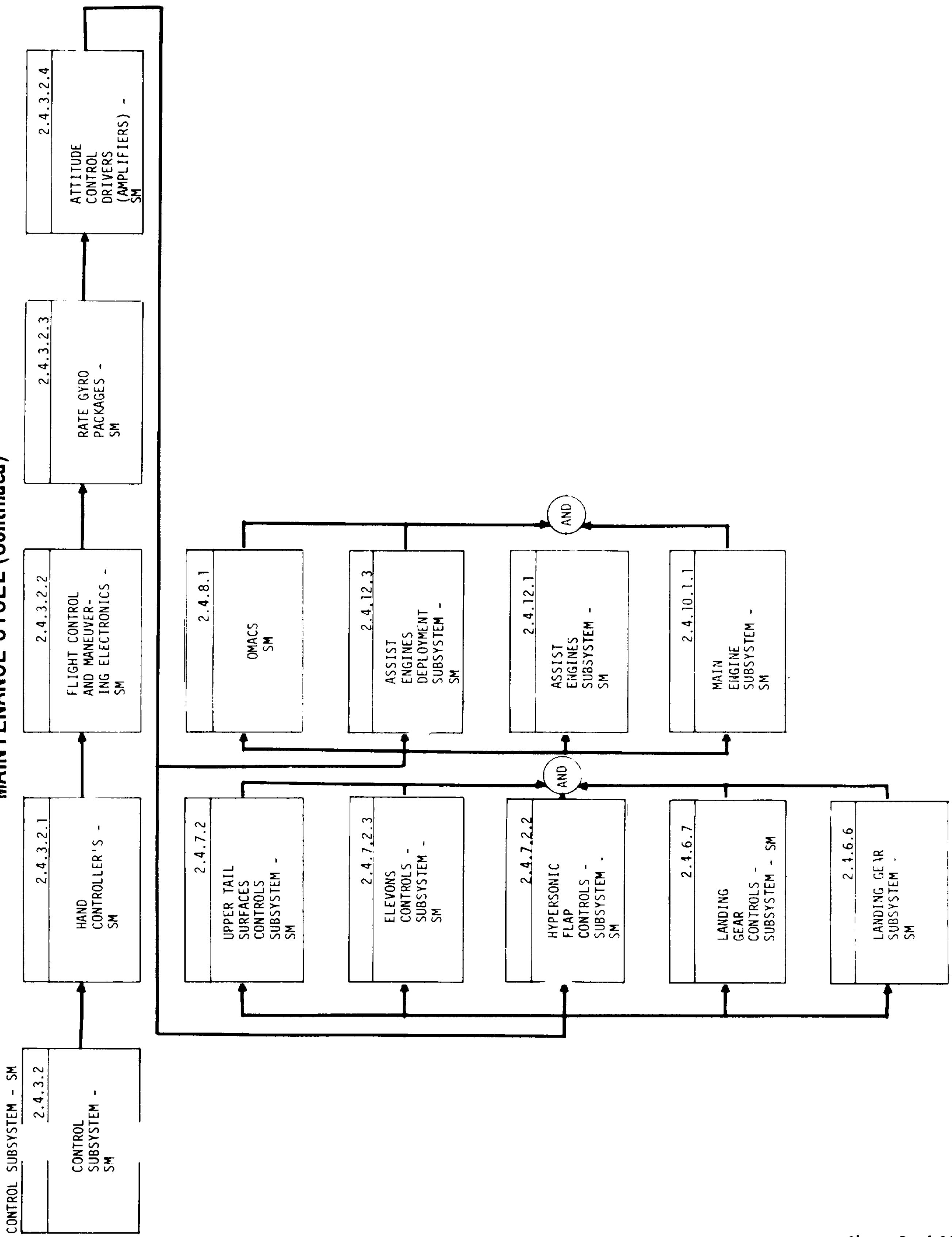

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Figure 4-6

MAINTENANCE CYCLE (Continued)

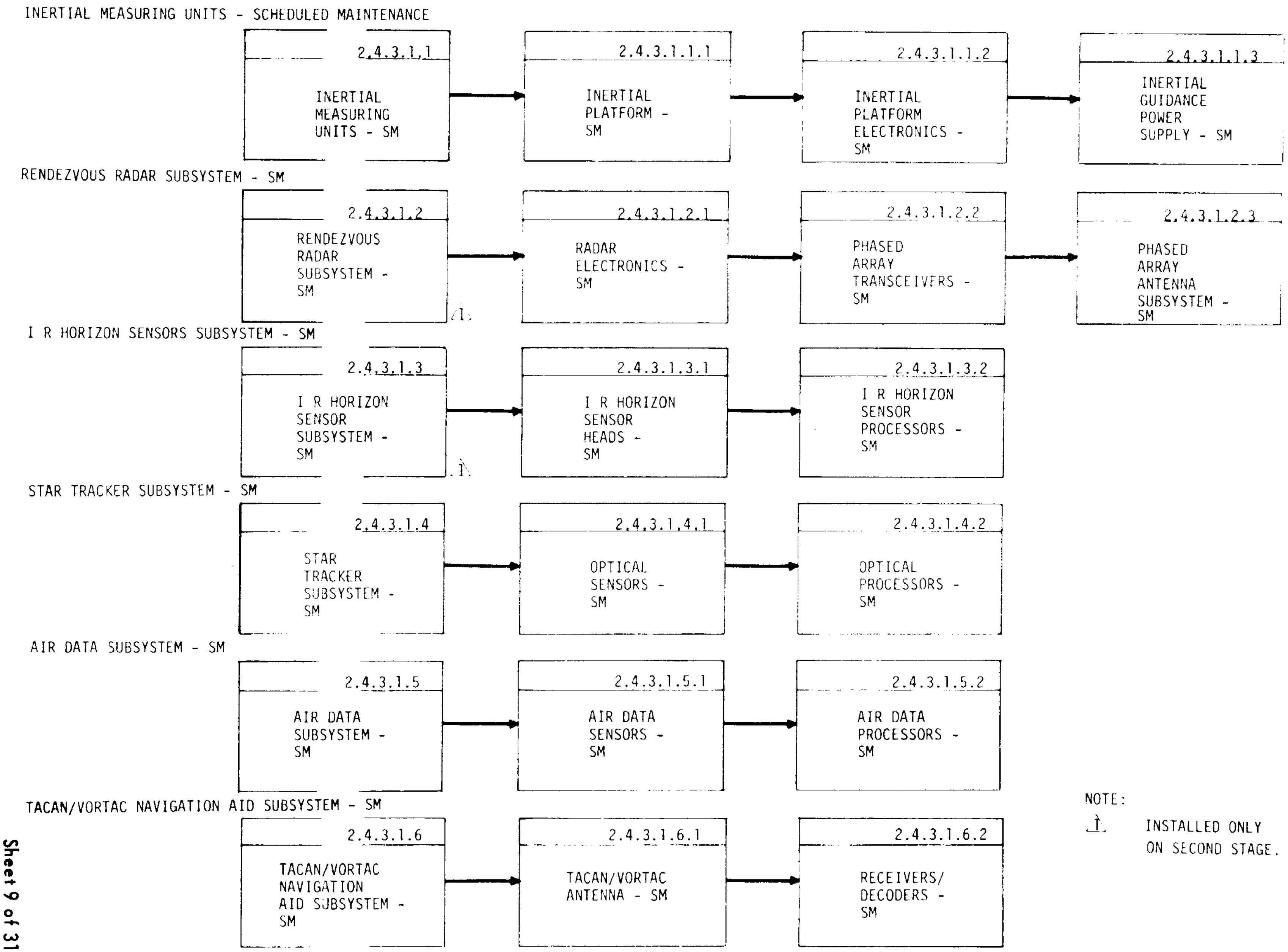

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Figure 4-6

MAINTENANCE CYCLE (Continued)

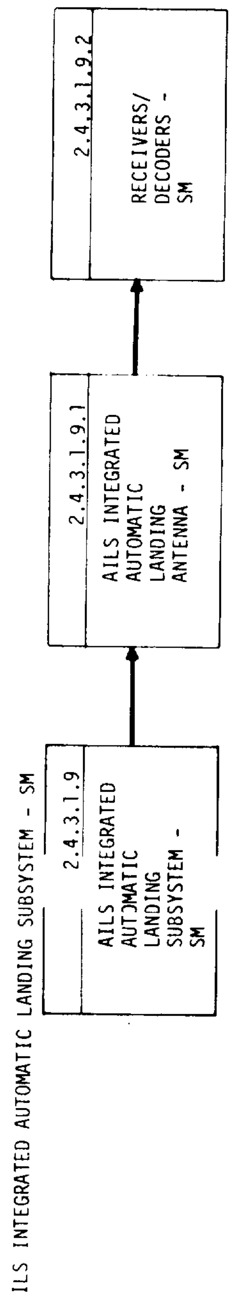

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Figure 4-6

MAINTENANCE CYCLE (Continued)

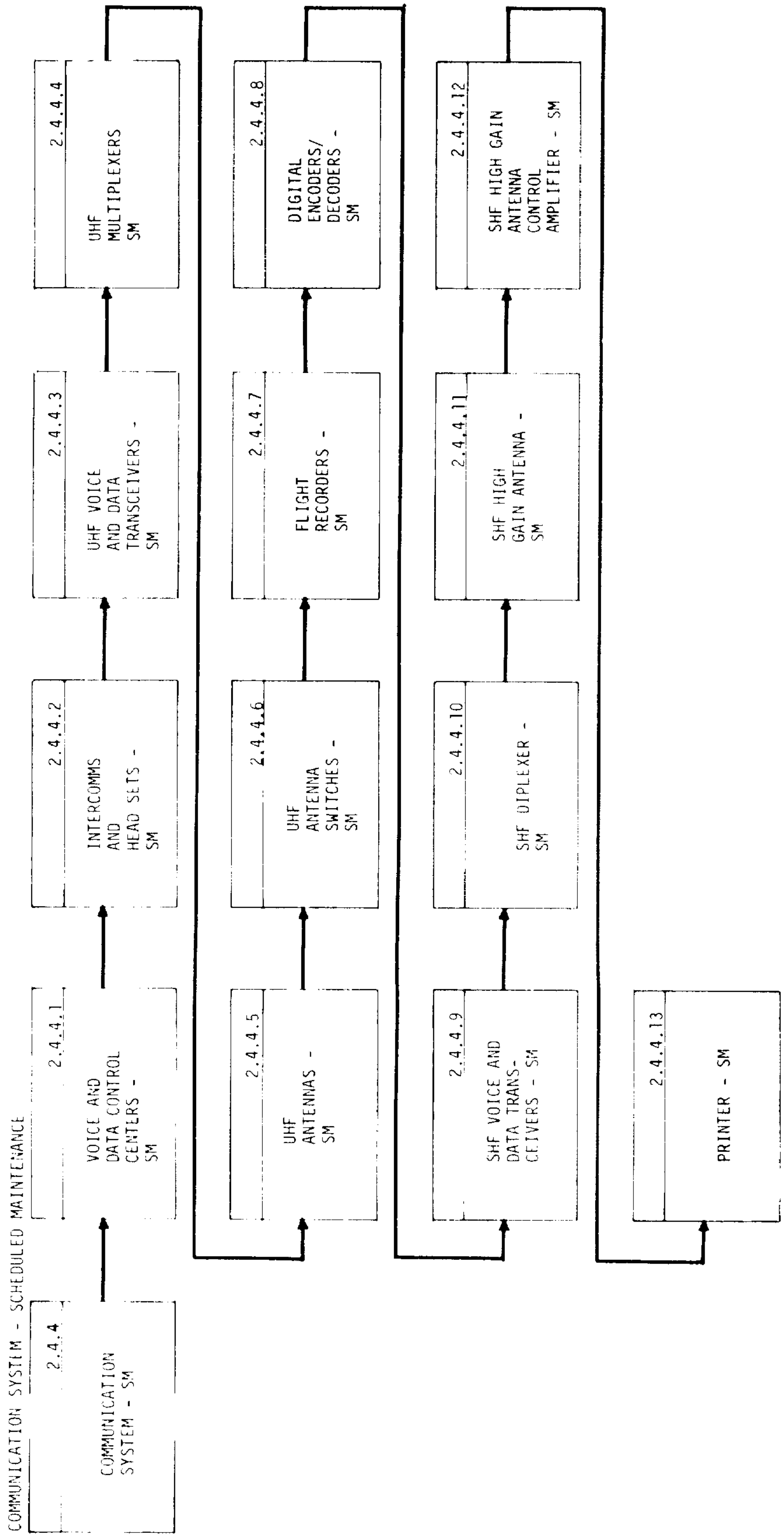

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Figure 4-6

MAINTENANCE CYCLE (Continued)

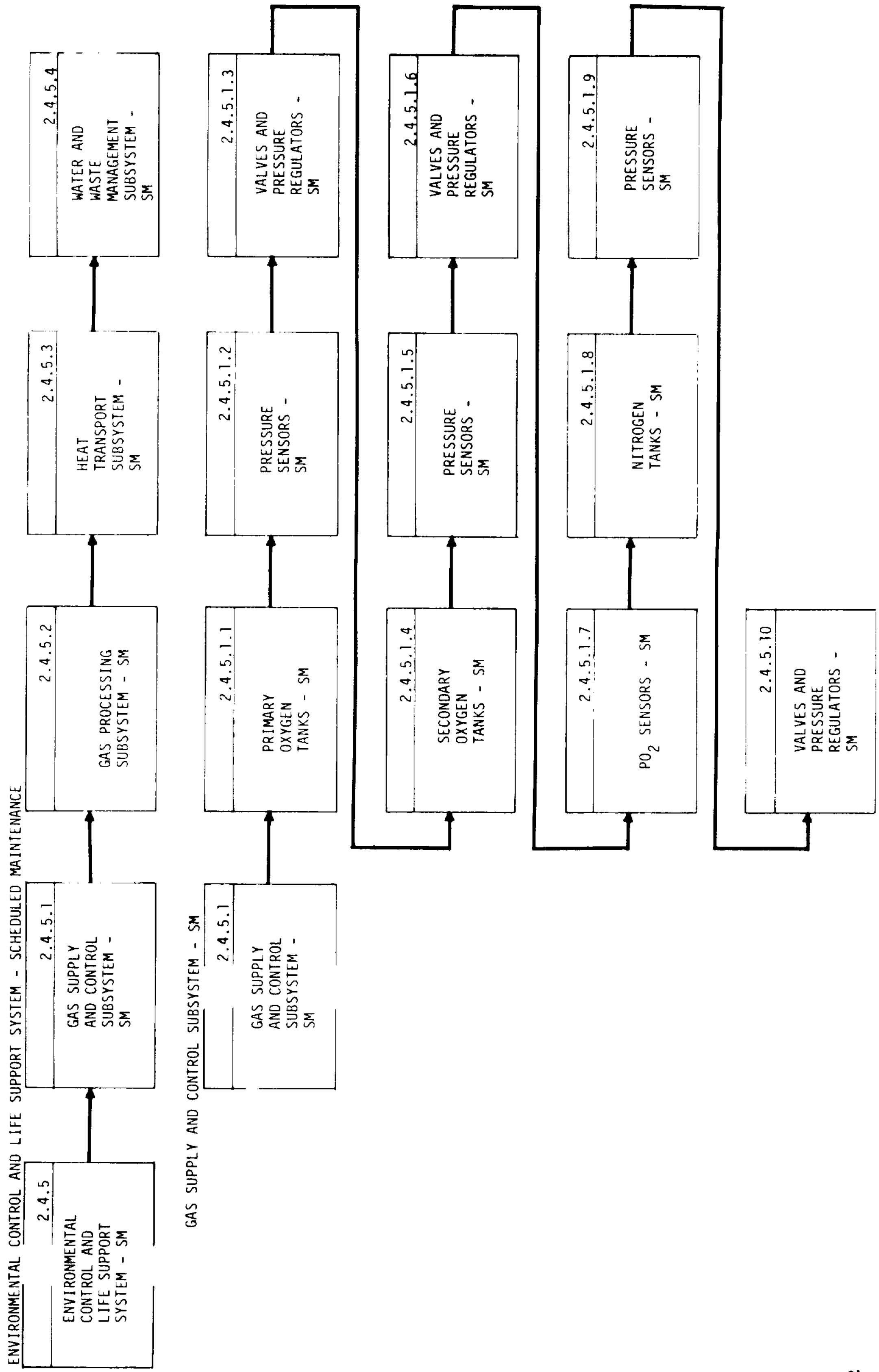

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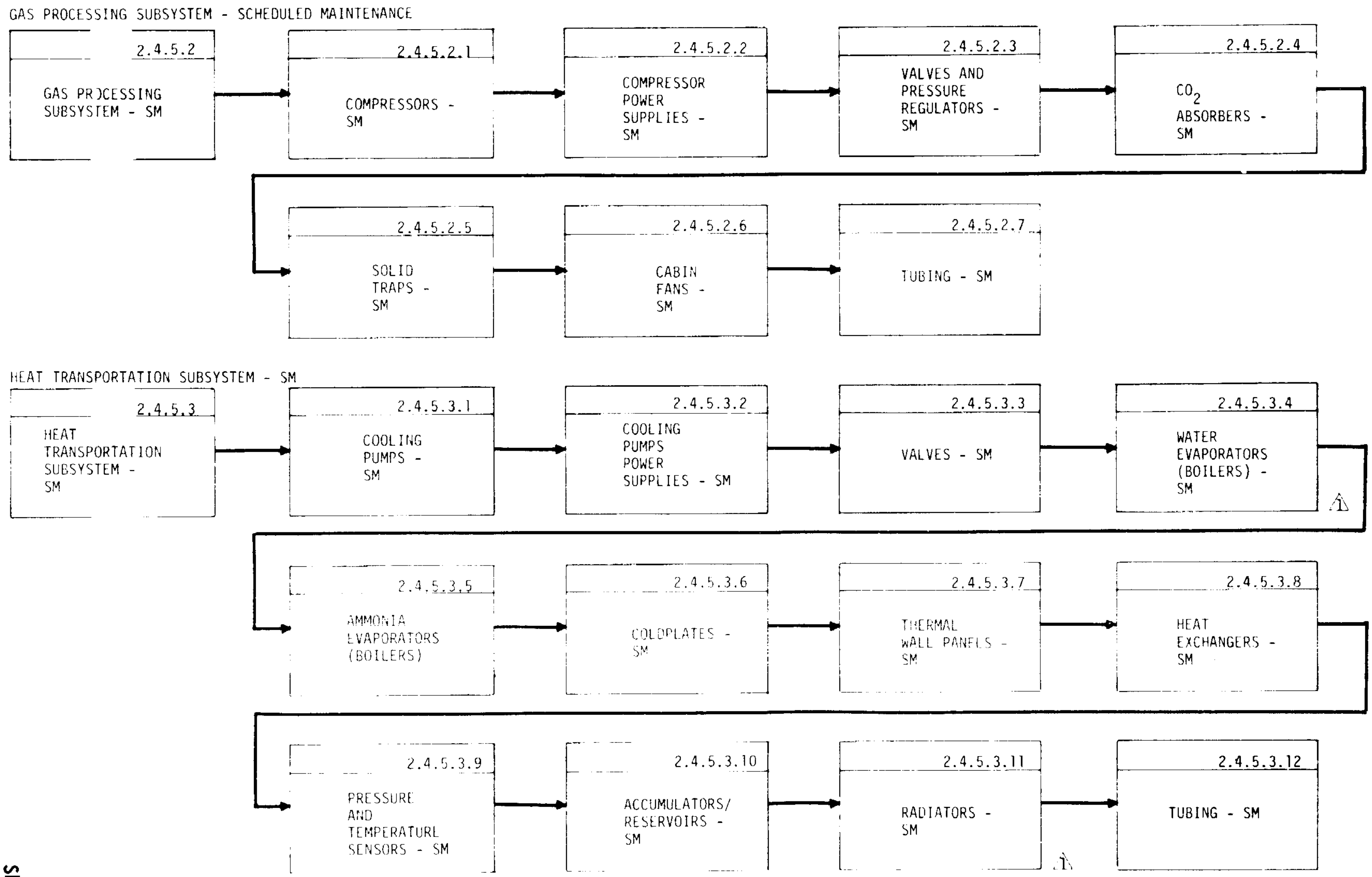
Figure 4-6

MAINTENANCE CYCLE (Continued)



NOTE:
1 INSTALLED ONLY ON SECOND STAGE.

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Figure 4-6

MAINTENANCE CYCLE (Continued)

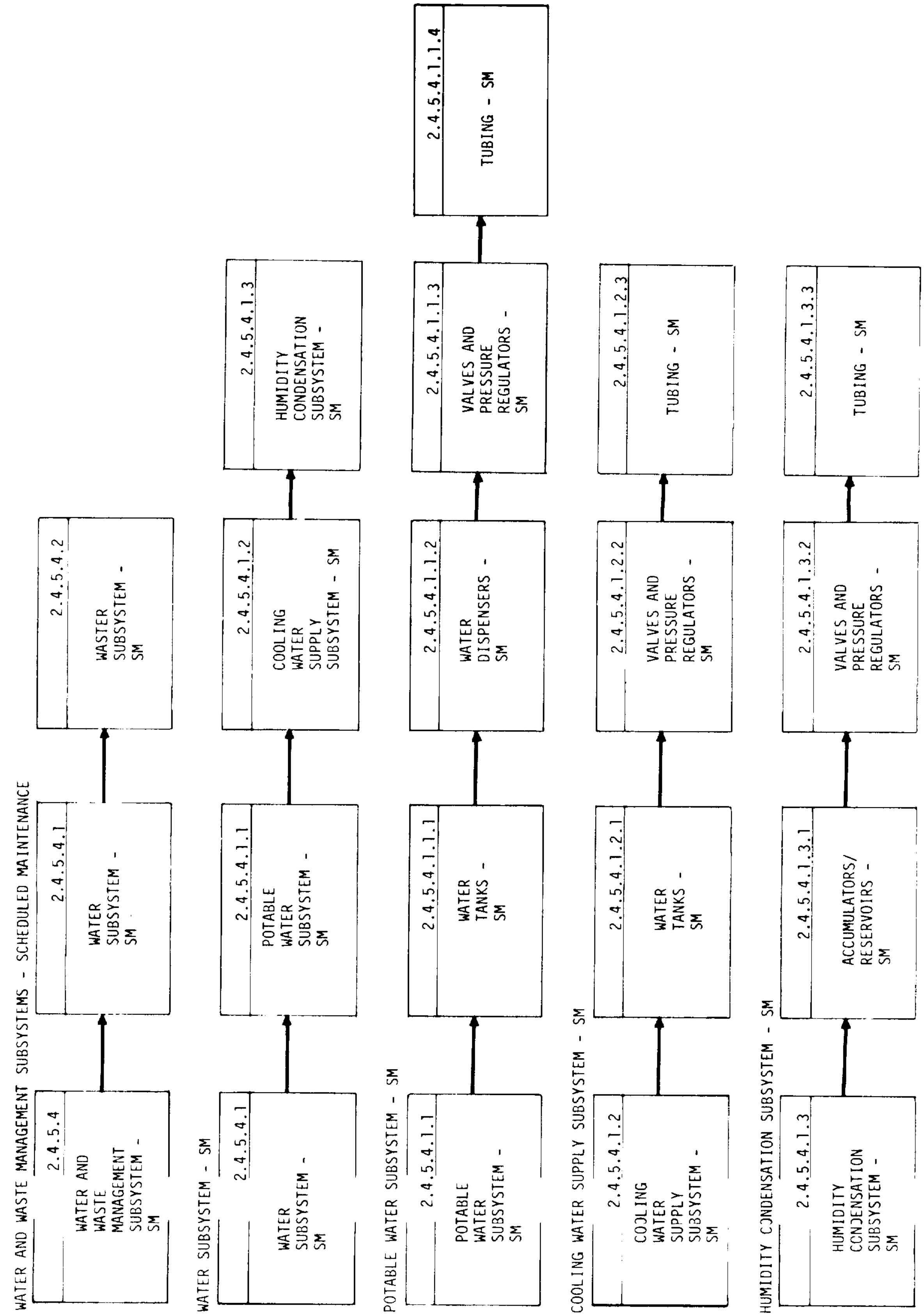

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Figure 4-6

MAINTENANCE CYCLE (Continued)

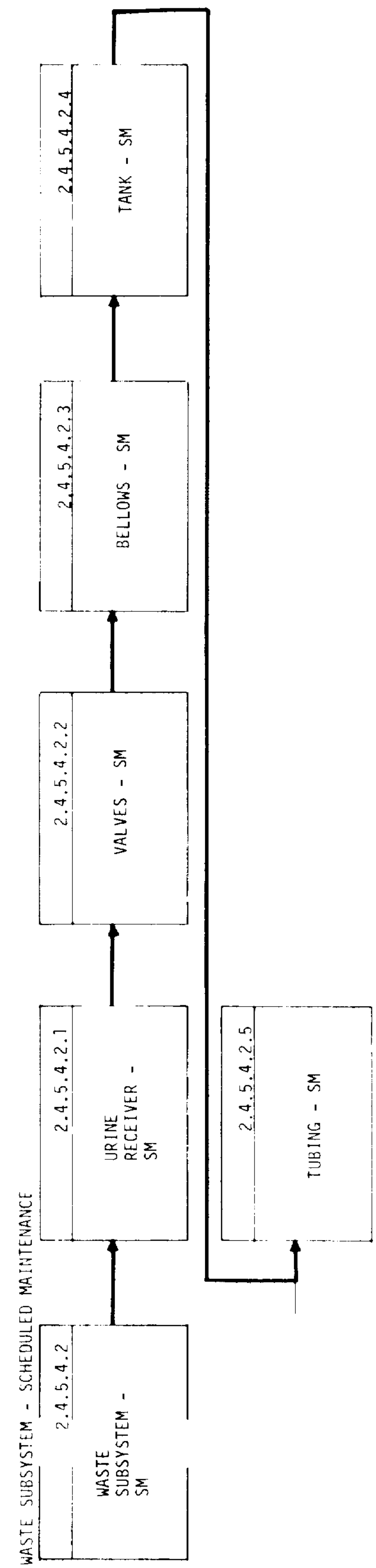

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Figure 4-6

MAINTENANCE CYCLE (Continued)

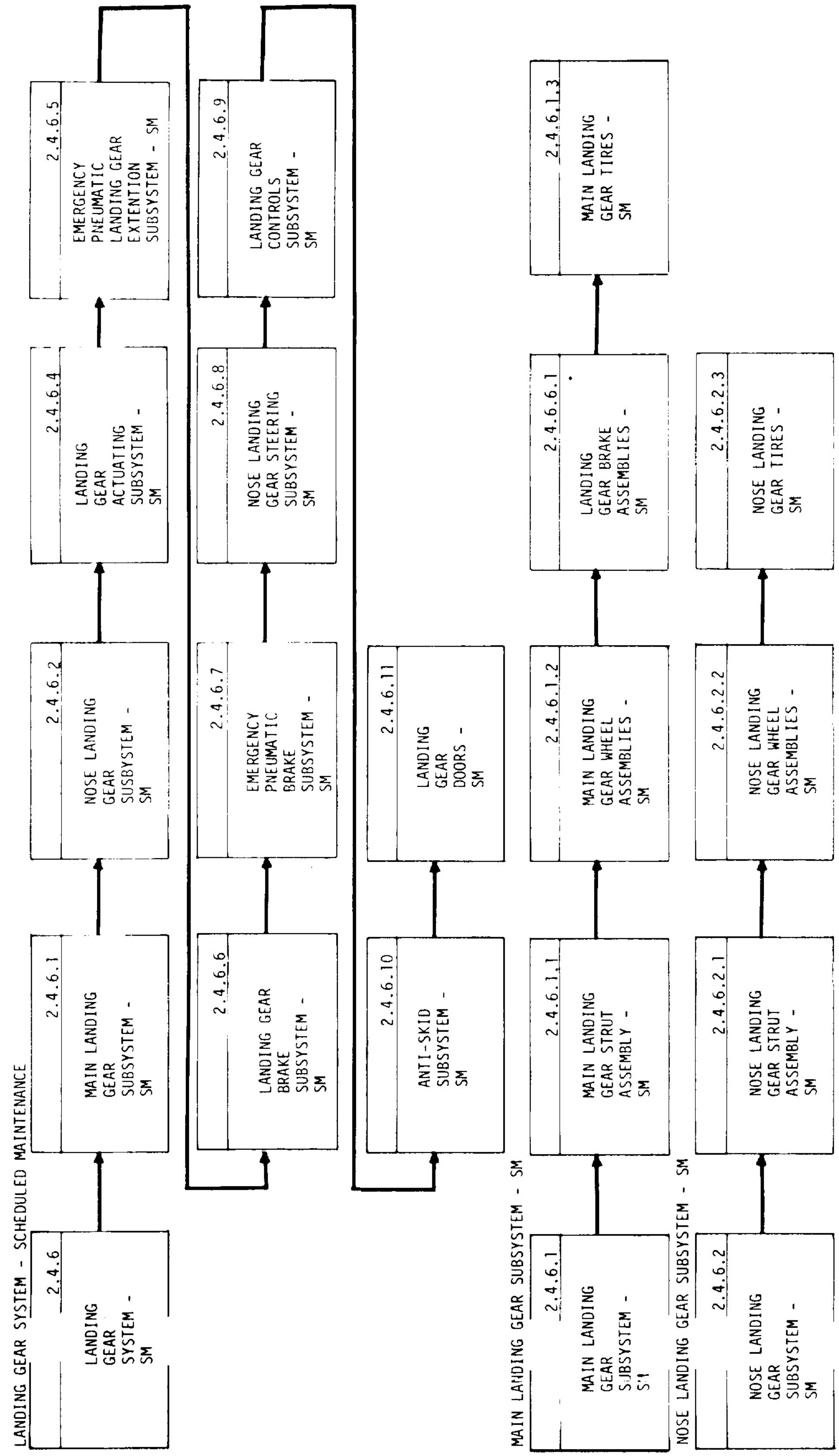

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Figure 4-6

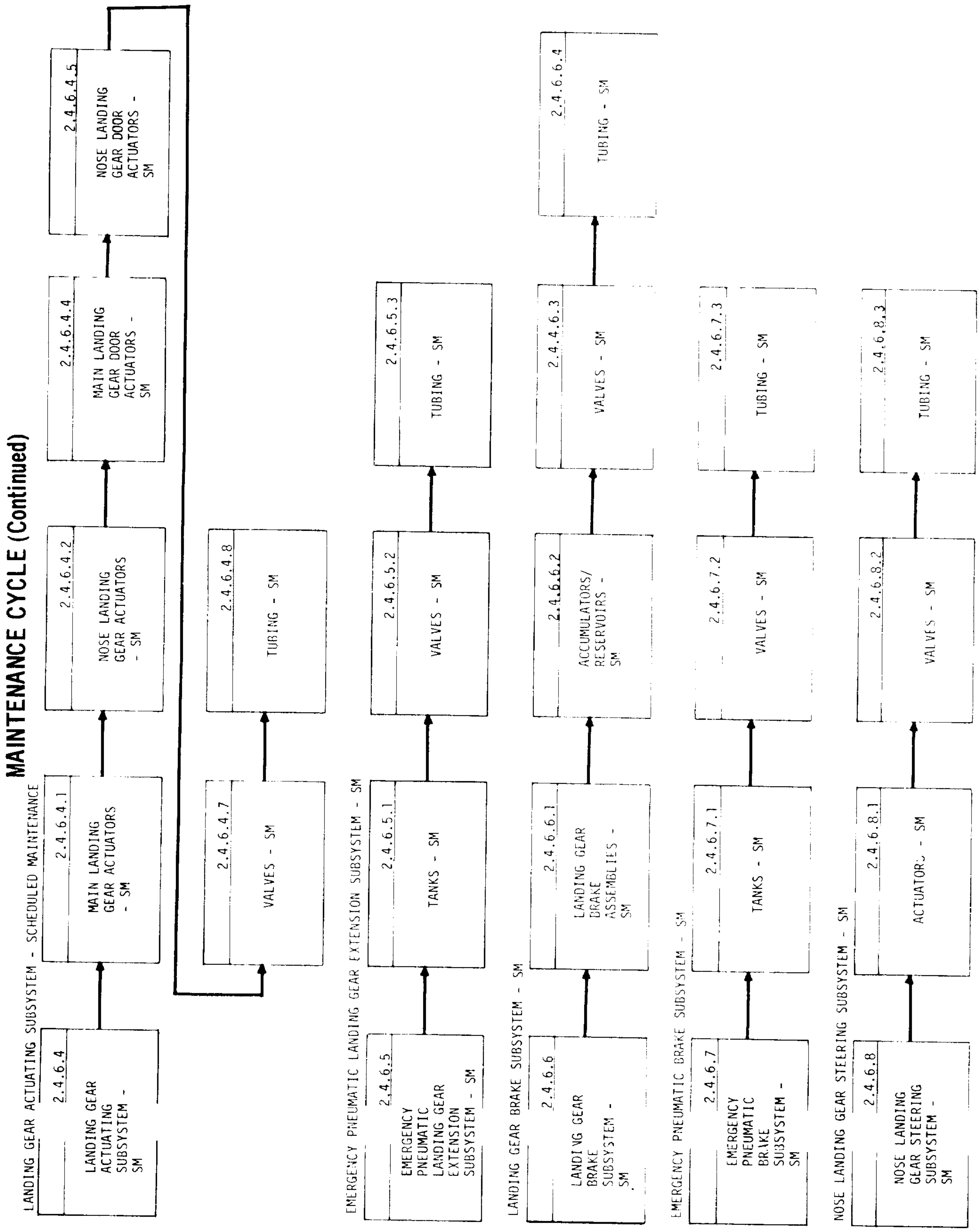

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Figure 4-6

MAINTENANCE CYCLE (Continued)

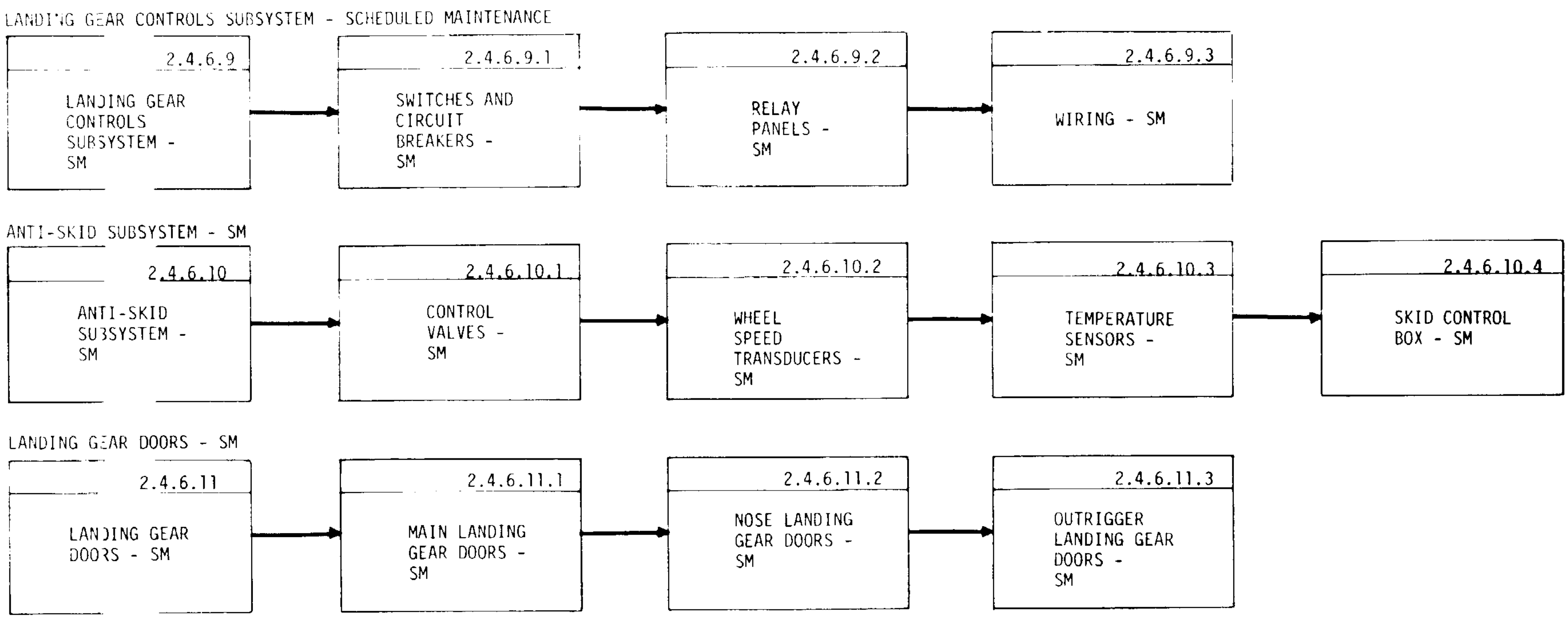

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Figure 4-6

MAINTENANCE CYCLE (Continued)

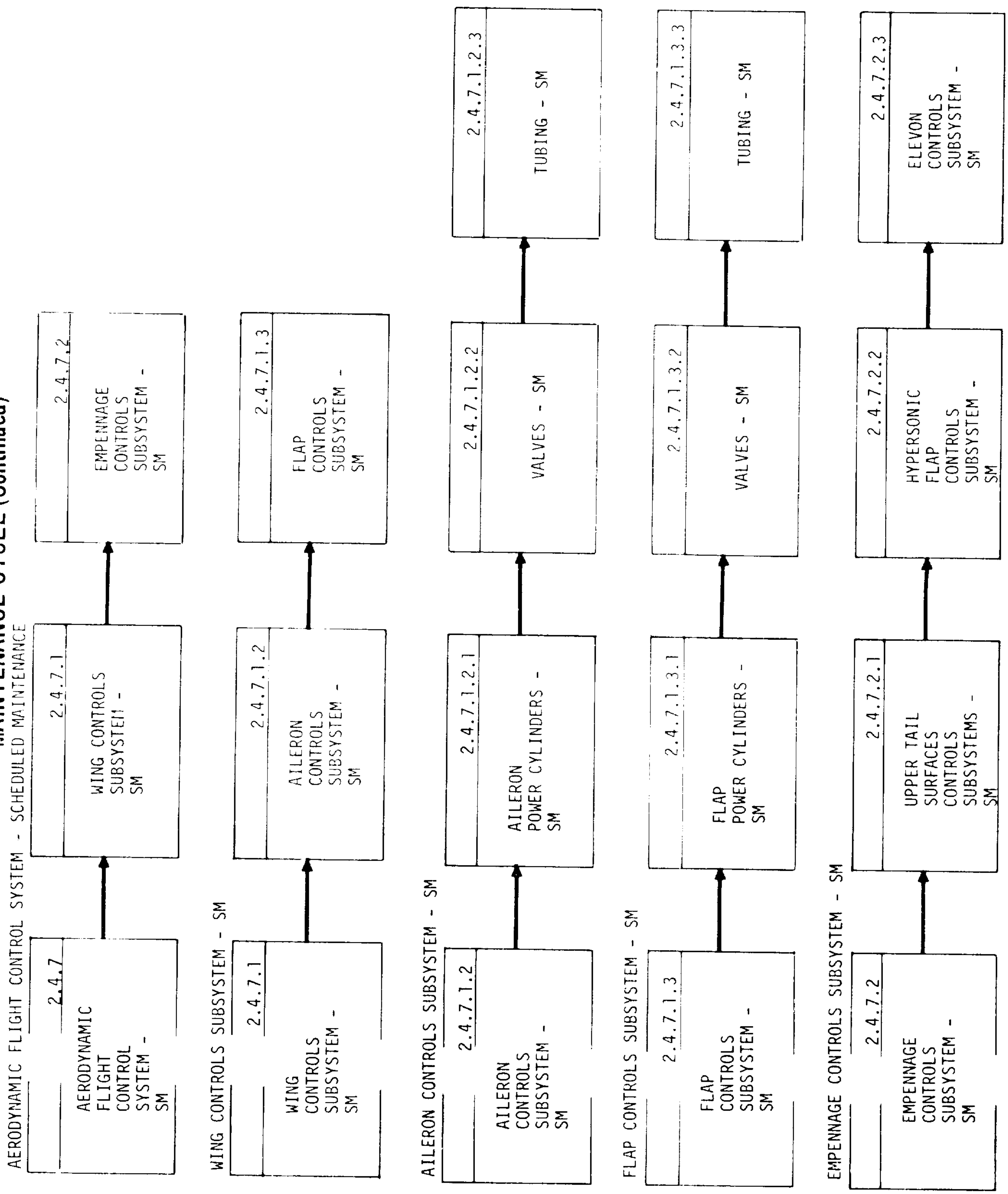

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Figure 4-6

MAINTENANCE CYCLE (Continued)

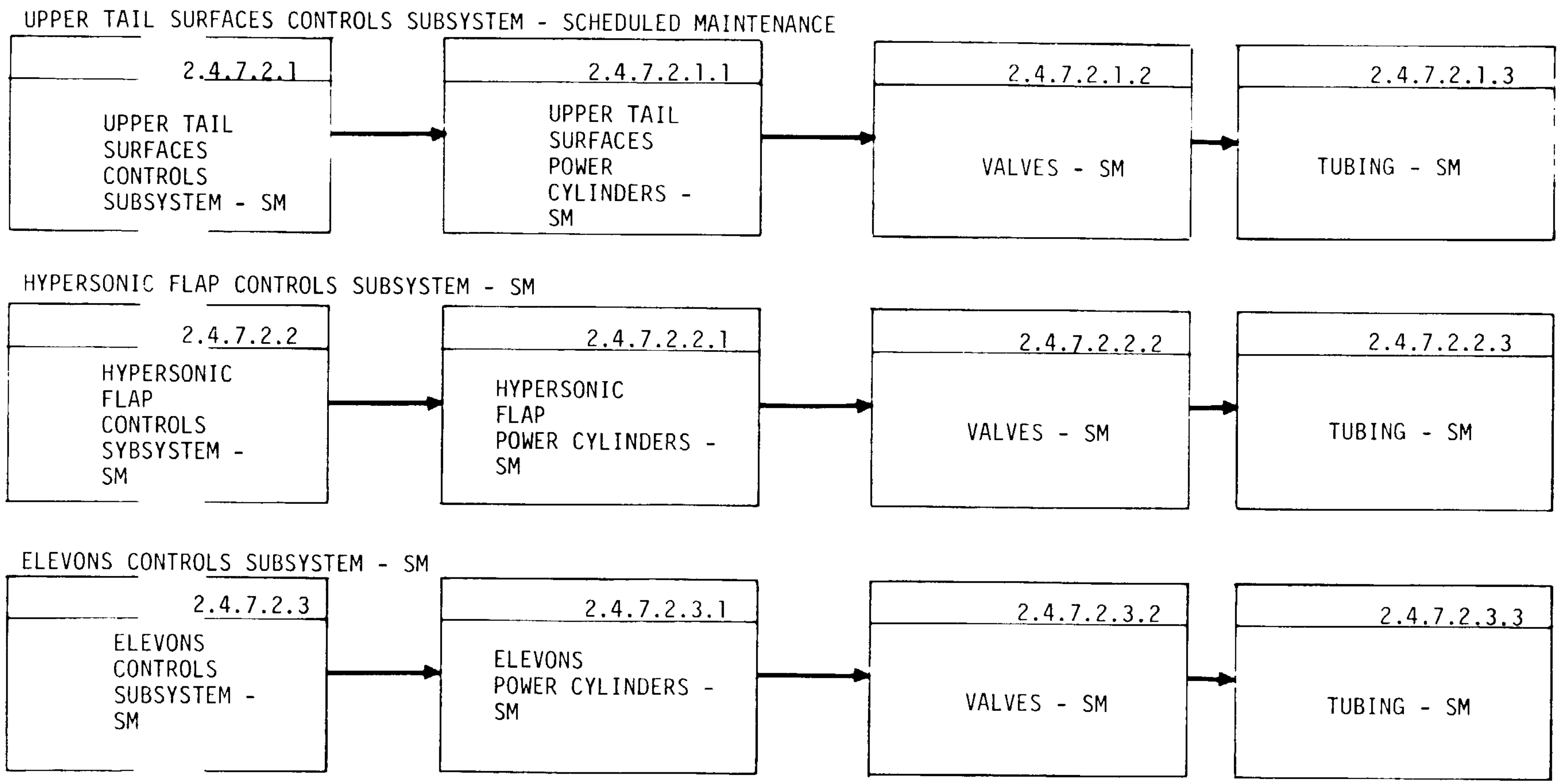

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Figure 4-6

MAINTENANCE CYCLE (Continued)

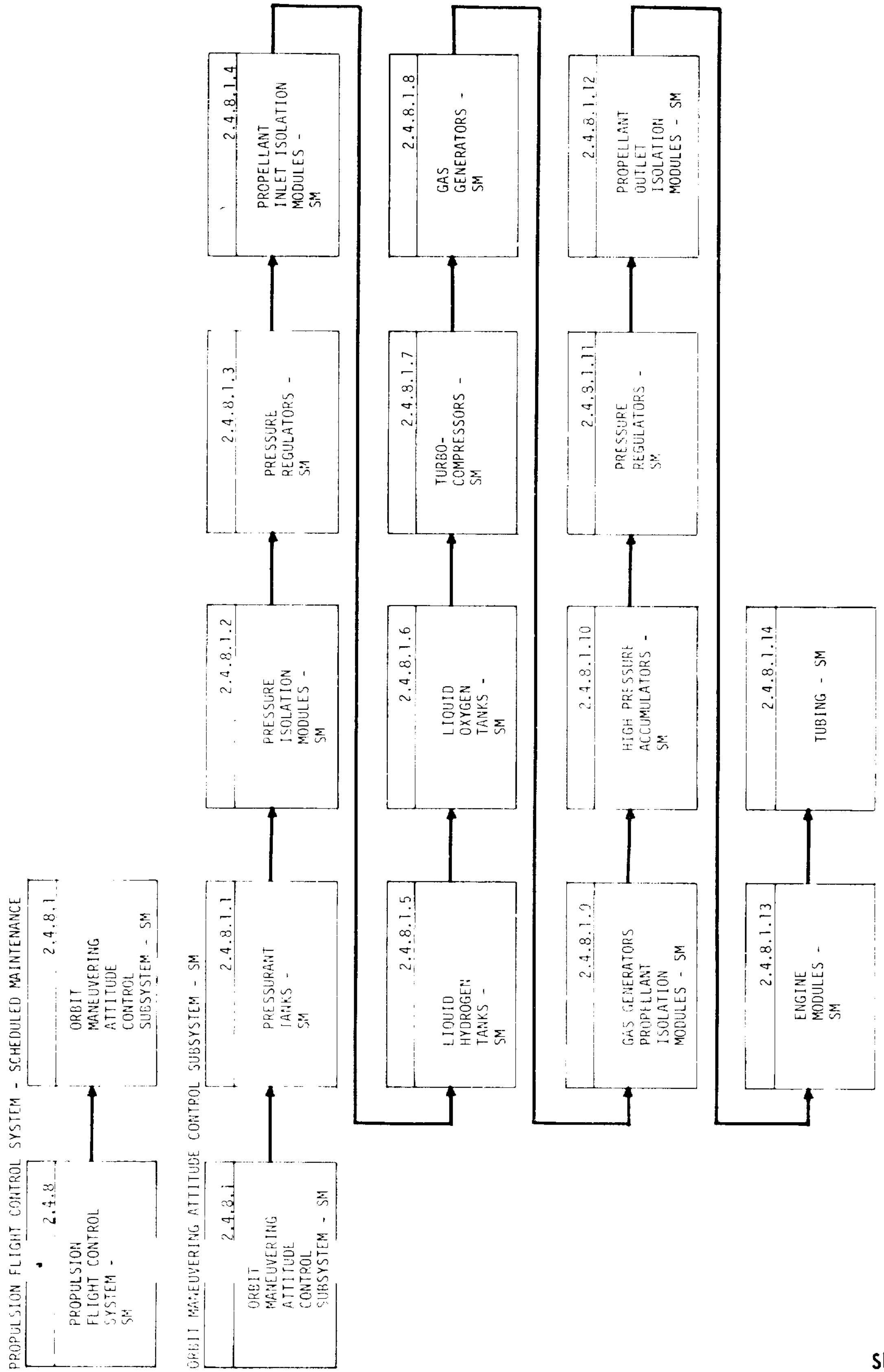

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Figure 4-6

MAINTENANCE CYCLE (Continued)

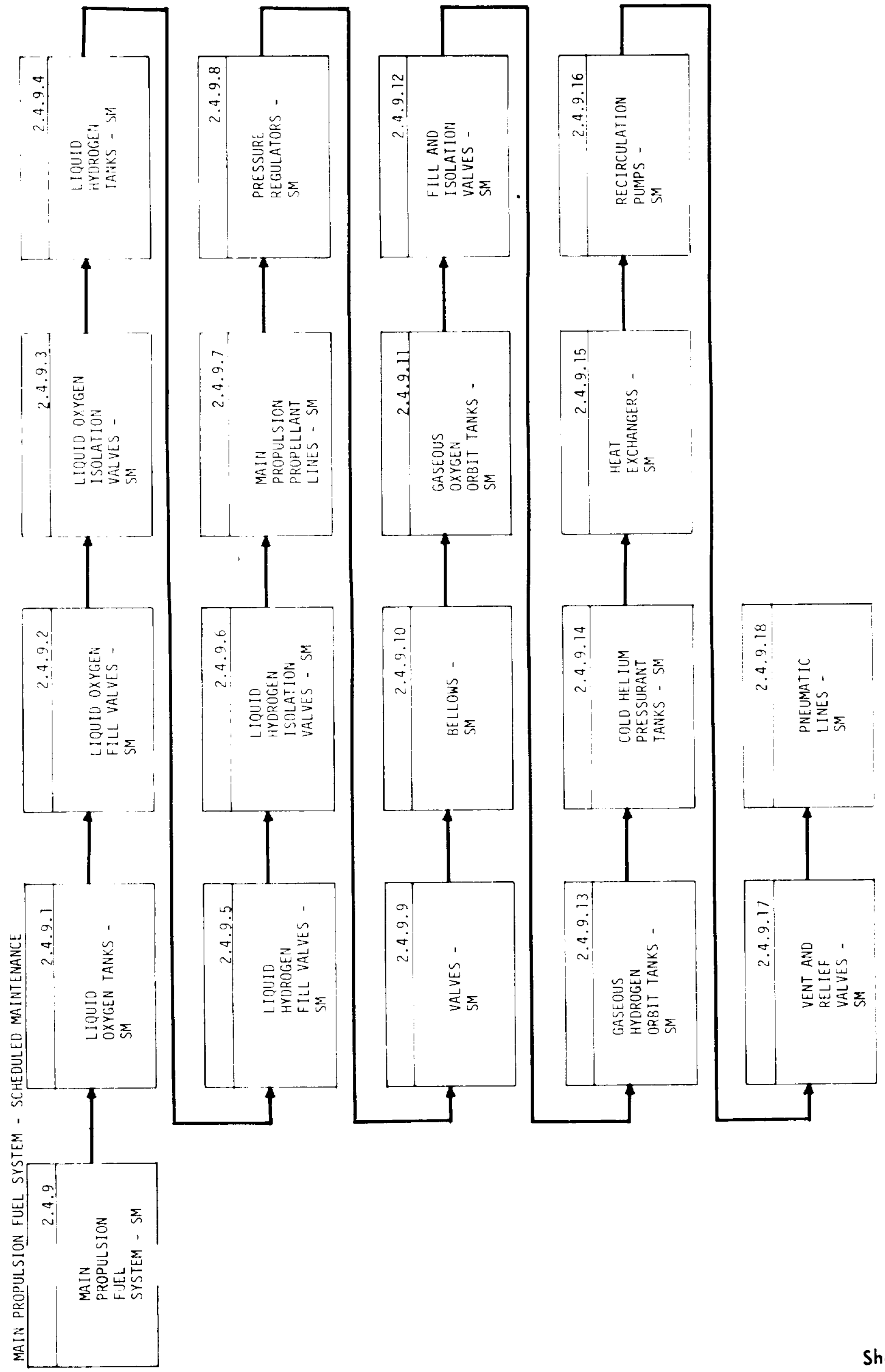

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Figure 4-6

MAINTENANCE CYCLE (Continued)

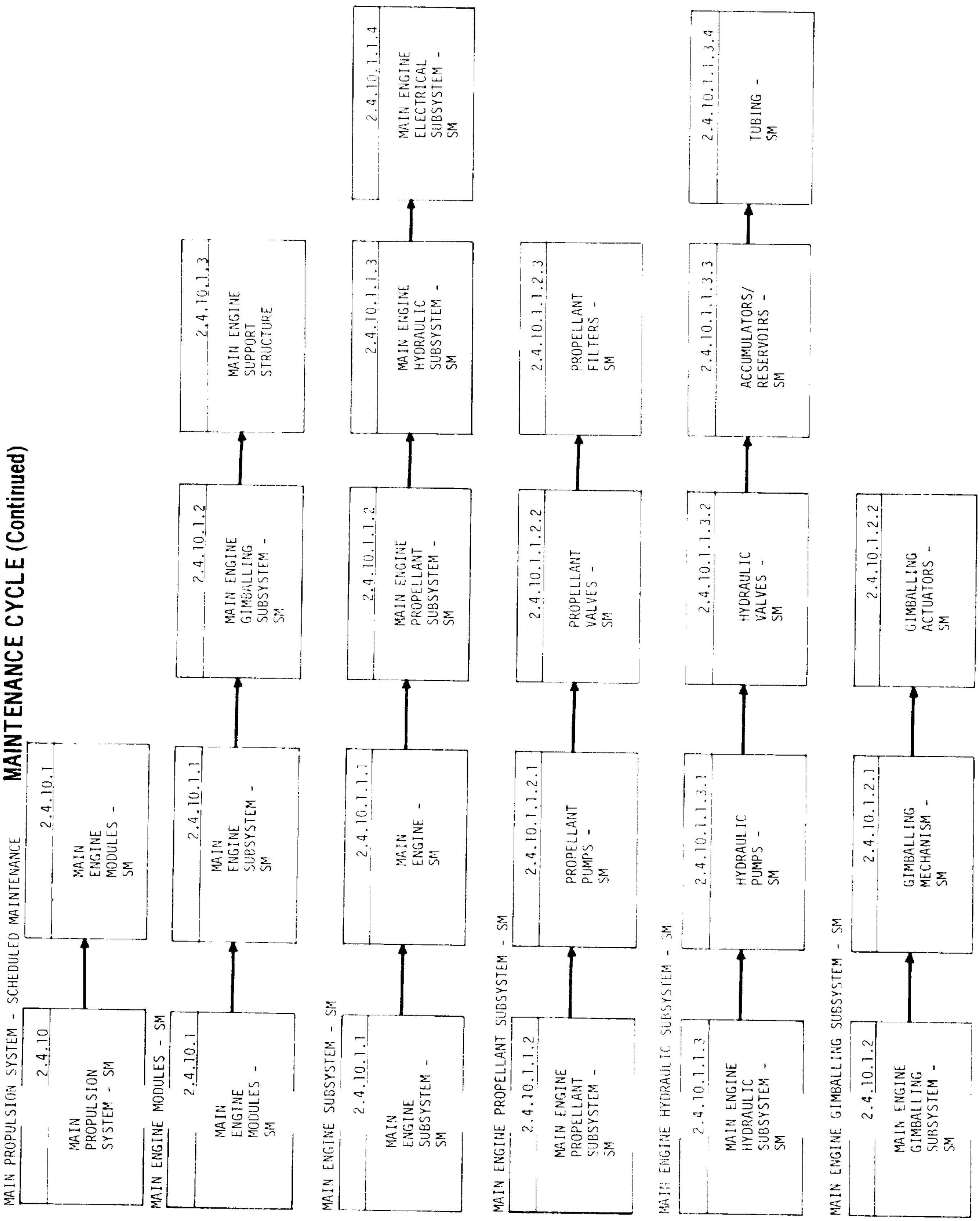

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Figure 4-6

MAINTENANCE CYCLE (Continued)

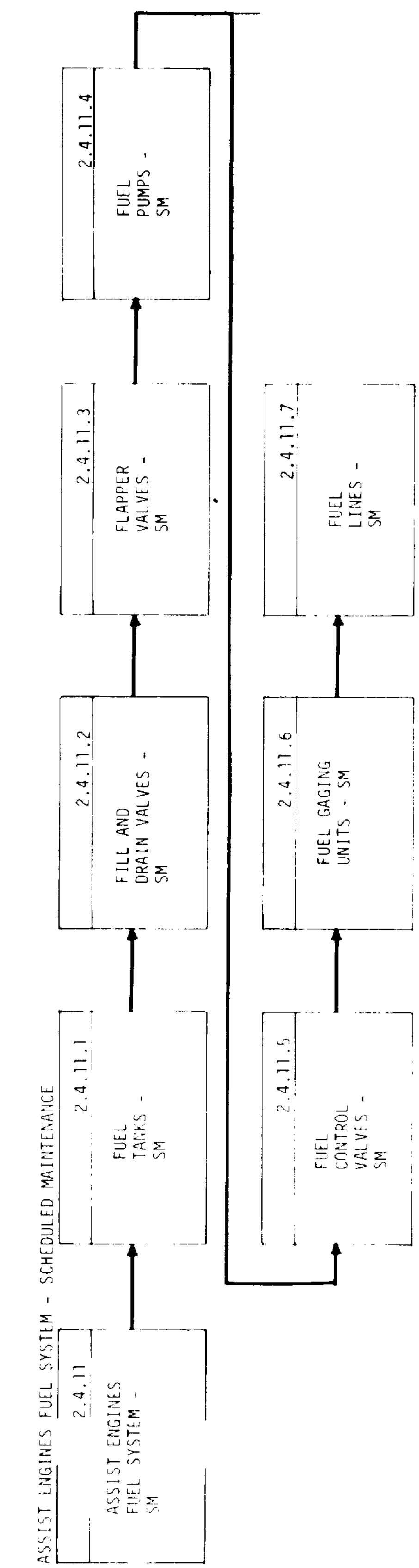

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Figure 4-6

MAINTENANCE CYCLE (Continued)

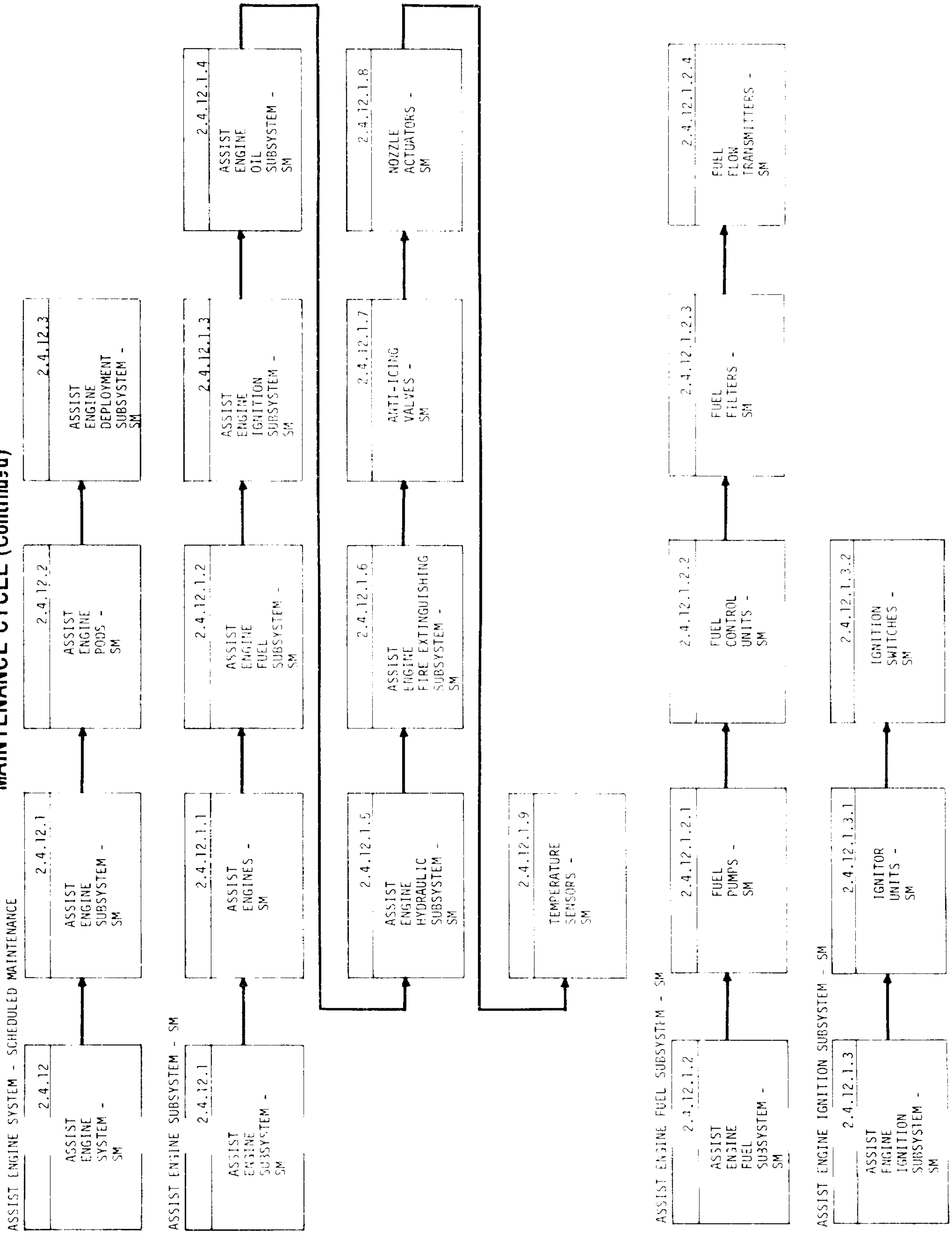

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Figure 4-6

MAINTENANCE CYCLE (Continued)

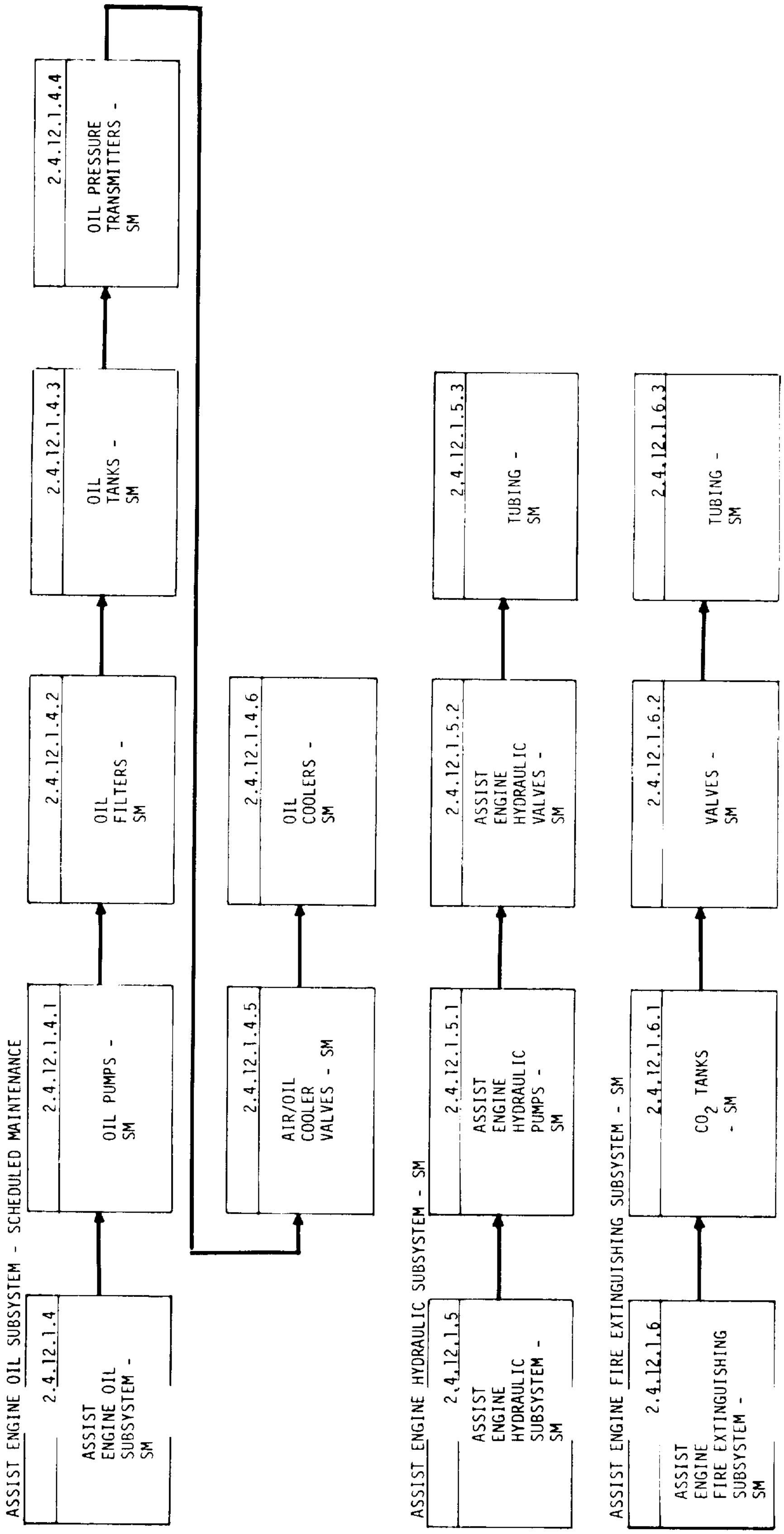

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Figure 4-6

MAINTENANCE CYCLE (Continued)

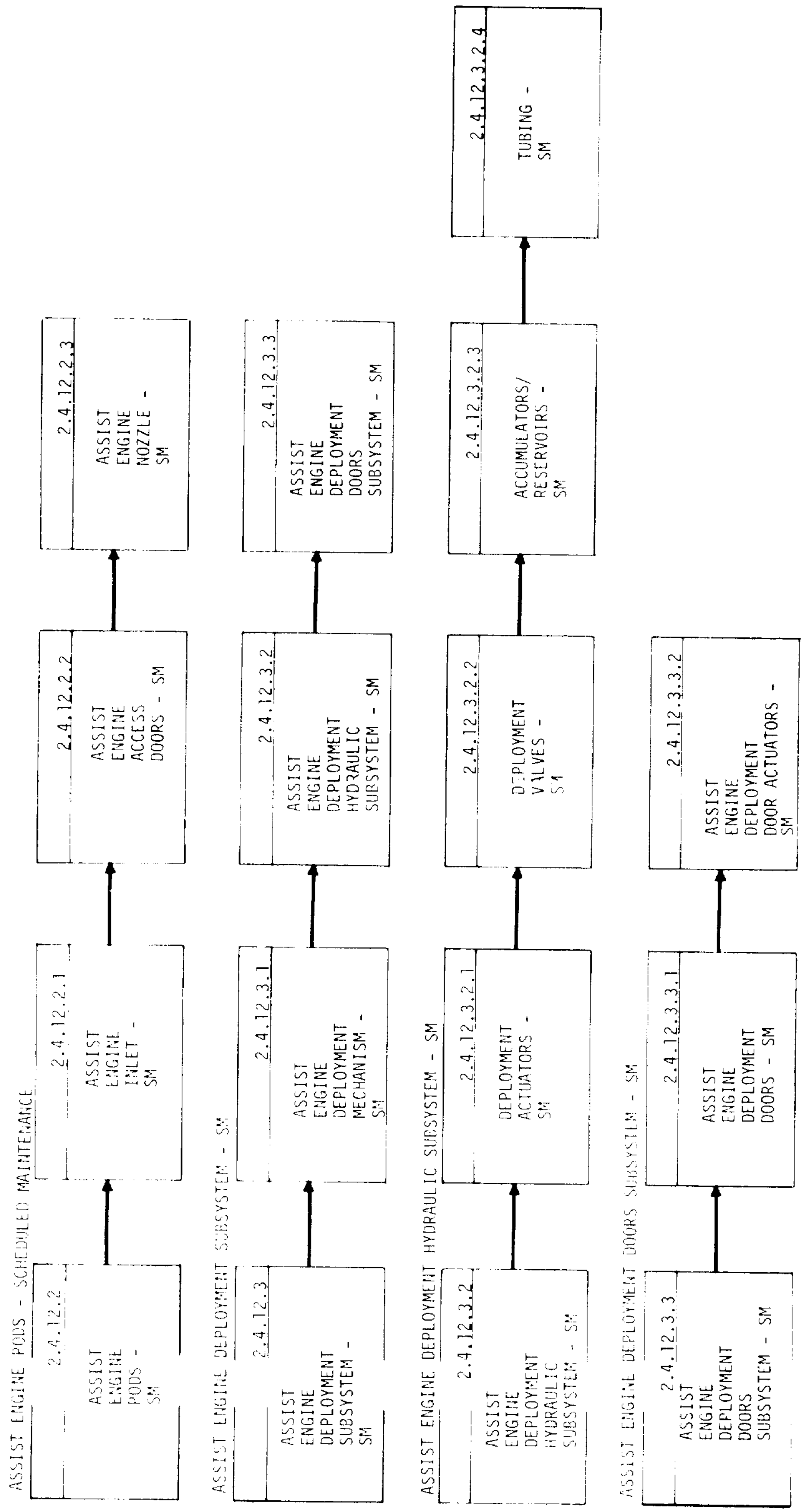

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Figure 4-6

MAINTENANCE CYCLE (Continued)

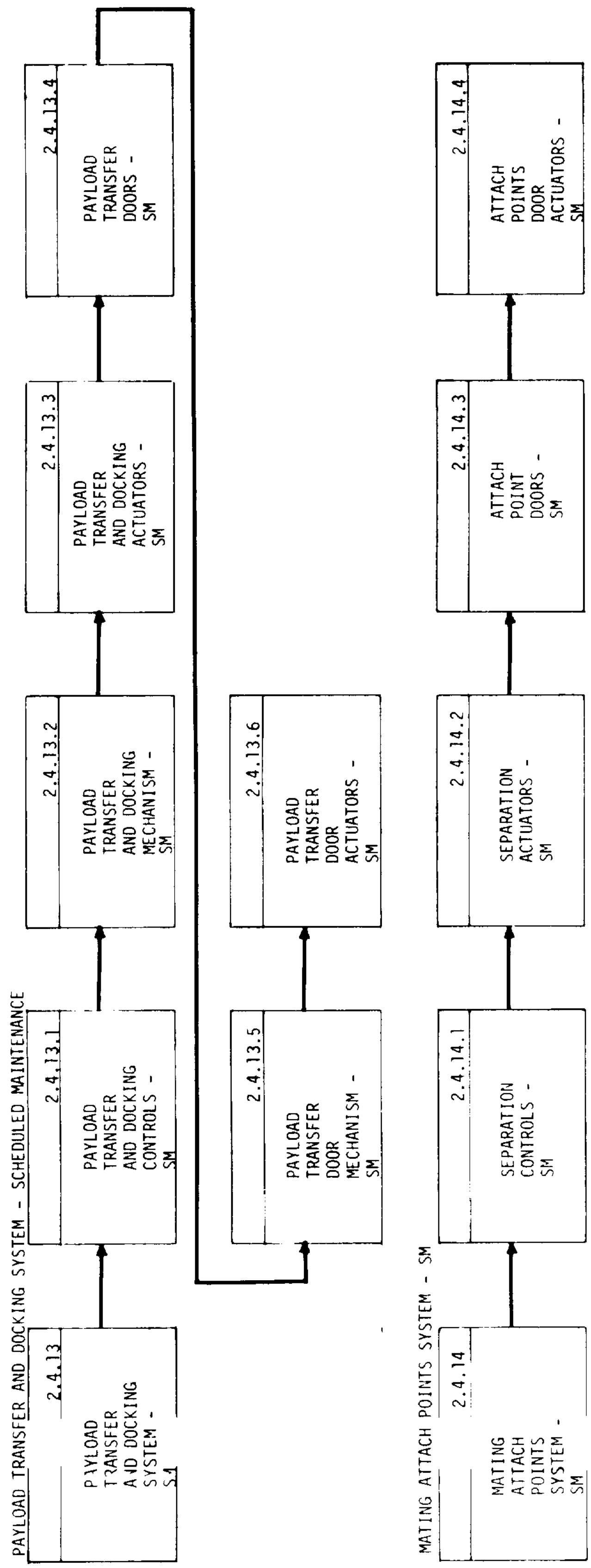

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Figure 4-6

MAINTENANCE CYCLE (Continued)

HEAT PROTECTION - SCHEDULED MAINTENANCE

- 2.4.15 HEAT PROTECTION - SM
- 2.4.15.1 HEAT SHIELD PANEL INSTALLATION - SM
- 2.4.15.2 INSULATION BLANKET INSTALLATION - SM

HEAT SHIELD PANEL INSTALLATION - SM

- 2.4.15.1 HEAT SHIELD PANEL INSTALLATION - SM
- 2.4.15.1.1 HEAT SHIELD PANELS - SM
- 2.4.15.1.2 HEAT SHIELD PANEL SUPPORT STRUCTURE - SM
- 2.4.15.1.3 HEAT SHIELD PANEL ATTACHING PARTS - SM

HEAT SHIELD PANELS - SM

- 2.4.15.1 HEAT SHIELD PANELS - SM
- 2.4.15.1.1 TO NICKEL HEAT SHIELD PANELS - SM
- 2.4.15.1.2 BORON-TITANIUM HEAT SHIELD PANELS - SM
- 2.4.15.1.3 BORON-ALUMINUM HEAT SHIELD PANELS - SM

HEAT SHIELD PANEL ATTACHING PARTS - SM

- 2.4.15.1.3 HEAT SHIELD PANEL ATTACHING PARTS - SM
- 2.4.15.1.3.1 ATTACHING TIE STRIPS - SM
- 2.4.15.1.3.2 SCREWS - SM
- 2.4.15.1.3.3 NUTS - SM

INSULATION BLANKET INSTALLATION - SM

- 2.4.15.2 INSULATION BLANKET INSTALLATION - SM
- 2.4.15.2.1 INSULATION BLANKET - SM
- 2.4.15.2.2 INSULATION BLANKET SUPPORT - SM

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Figure 4-6

MAINTENANCE CYCLE (Continued)

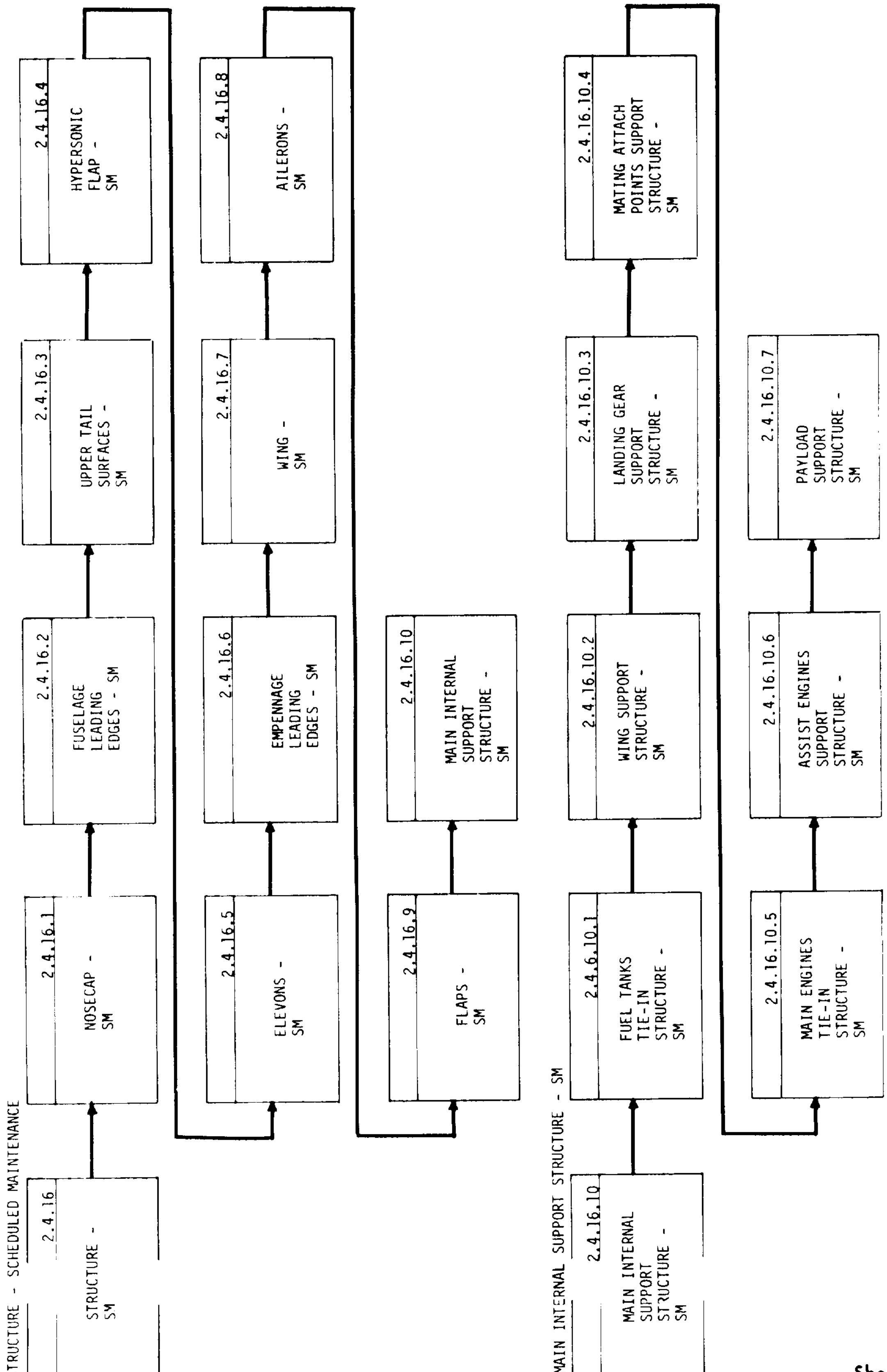

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Figure 4-6

MAINTENANCE CYCLE (Continued)

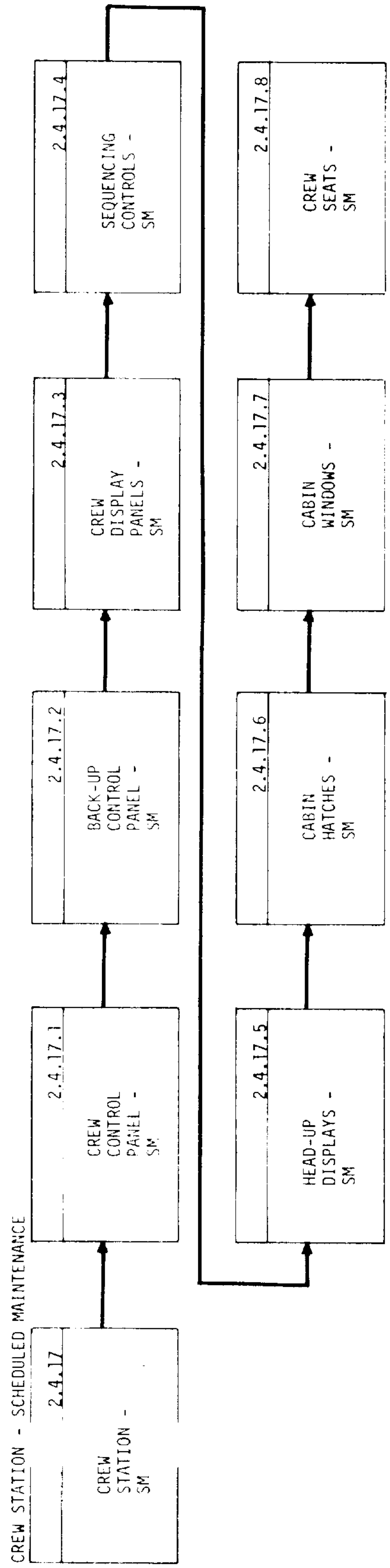

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Figure 4-6

To arrive at the most realistic tasks requirements and times to complete each task, some subsystem have been taken down to the fifth level of functional flow diagraming. In conjunction with this analysis, commercial airline practices were evaluated to determine the types and quantity of maintenance that could be expected after every mission.

This airline data indicated that during periods of scheduled maintenance, 50 to 90% of the total labor expended is for unscheduled corrective maintenance. The analysis of commercial airline data suggested that the 50% factor of unscheduled maintenance would apply to the ILRV-LRC vehicle.

First level maintenance functional actions are illustrated on Sheet 1 of 31 of Figure 4-6. As indicated by this diagram, performance of quality assurance inspection is a parallel action to all maintenance functions. Functional actions required to provide access for maintenance are shown on Sheets 2 and 3 of Figure 4-6. Vehicle systems on which scheduled maintenance is performed are presented on Sheet 4 of Figure 4-6. Performance of scheduled maintenance on the vehicle system and subsystems, and their major components and assemblies are presented on Sheets 5 through 31 of Figure 4-6.

<u>Task Analysis</u> - The task analysis, as presented in Table 4-4 lists the functions, tasks, and subtasks to be performed during the maintenance cycle. Each task or subtask was analyzed to estimate the frequency of occurrence, (i.e., after every flight (AEF) or after an elapsed time in hours). The analysis also included an estimate of the number of manhours and personnel and the elapsed time required to complete each task or subtask. The tasks were divided into two parts, visual inspection requirements and functional requirements. The task analyses were based on the functional performance consisting mostly of visual inspections of the Maintenance Cycle illustrated in the functional flow block diagram of Figure 4-6.

<u>Maintenance Cycle Inspections</u> - The maintenance cycle inspections differ for each turnaround. These types of inspections are dependent on number of missions elapsed times, or number of cycles exercised on the equipment and are explained in Table 4-5.

<u>Time Analysis</u> - The timeline analysis shown in Figures 4-7 and 4-8 are the product of the preceding functional flow block diagrams and task anslysis. Timeline analysis of the visual inspections and functional checks are shown in

TASK ANALYSIS

Table 4-4

SCHEDULED MAINTENANCE (SM)

FUNCTION & /TASK/ SUB-TASK NUMBER	FREQUENCY	M/H'S - M/M'S	PERSONAL	ELAPSED TIME
ONBOARD CHECKOUT SYSTEM (SM) 2.4.1		4.0 M/H'S	-	1.75 HR.
. VISUAL	AEF	40 M/M'S	-	30 MIN.
. INSP. ALL SWITCHES FOR SECURITY OF MOUNTING AND FUSE AND CIRCUIT BREAKERS FOR BLOWN OR UNLATCHED CONDITION.		30 M/M'S	2	15 MIN.
. INSP. ALL VISIBLE WIRE BUNDLES FOR CHAFING, LOOSE OR BROKEN TIES AND ANCHOR CLIPS AND LOOSE OR CORRODED TERMINALS.		2 M/H'S	4	30 MIN.
. INSP. ALL INDICATORS FOR DAMAGE AND SECURITY.		10 M/M'S	1	10 MIN.
. INSP. THE FOLLOWING EQUIPMENT FOR SECURITY OF MOUNTING: . DATA INTERCHANGE AND CONTROL UNIT . DCS CONTROL AND DISPLAY UNIT . CENTRAL MANAGEMENT COMPUTER		1 M/H	2	30 MIN.
. FUNCTIONAL	AEF	20 M/M'S	-	20 MIN.
. PERFORM A SELF TEST OPERATIONAL AND VERIFY PROPER READOUT		20 M/M'S	1	20 MIN.

NOTE: 1. AFTER EVERY FLIGHT (AEF)

Table 4-4
TASK ANALYSIS (Continued)

SCHEDULED MAINTENANCE (SM)

FUNCTION & /TASK/ SUB-TASK NUMBER	FREQUENCY	M/H'S - M/M'S	PERSONAL	ELAPSED TIME
ELECTRICAL POWER SYSTEM (SM) 2.4.2	AEF	19.0 M/H'S	-	4.3 HRS.
. VISUAL	AEF	4.2 M/H'S	-	1.0 HR.
. INSP. ALL SWITCHES FOR SECURITY OF MOUNTING AND FUSE AND CIRCUIT BREAKERS FOR BLOWN OR UNLATCHED CONDITION.		Table 4-41	4	15 MIN.
. INSP. ALL VISIBLE WIRE BUNDLES FOR CHAFING, LOOSE OR BROKEN TIES AND ANCHOR CLIPS AND LOOSE OR CORRODED TERMINALS.		4.0 M/H'S	4	1 HR.
. INSP. ELECTRICAL CONNECTOR PLUGS AT ALL COMPONENTS AND BULKHEADS FOR SECURITY, CRACKS, CORRODED OUTER SHELL, AND FRAYED WIRES.		1.0 M/H	4	15 MIN.
. INSP. ALL TERMINALS FOR PROPER INSTALLATION AND SECURITY.		30.0 M/M'S	3	10 MIN.
. INSP. THE FOLLOWING EQUIPMENT FOR SECURITY OF MOUNTING:		2.2 M/H'S	1	12 MIN.
. BATTERIES (36 FOR 2ND STAGE) (4 FOR 1ST STAGE)				
. APU COMPONENTS				
. RELAY PANELS				
. LIGHT ASSEMBLIES				
. INSP. BATTERY AND ADJACENT STRUCTURE FOR CORROSION.		20 M/M'S	2	10 MIN.
. INSP. APU FOR LEAKS		15 M/M'S	3	5 MIN.
. INSP. FOR HOT SPOTS ON APU ENGINE CASE		15 M/M'S	3	5 MIN.
. INSP. APU FUEL LEVEL		2 M/M'S	1	2 MIN.
. INSP. ALL POWER SYSTEM MONITORS FOR PROPER CONDITION.		2 M/M'S	1	2 MIN.
. FUNCTIONAL		9.18 M/H'S	-	1.9 HR.
. CLEAN AND/OR REPLACE APU FUEL FILTERS		45 M/M'S	3	15 MIN.
. PERFORM APU OPERATIONAL, VERIFY READOUT AND LEAK CHECK.		40 M/M'S	4	10 MIN.

Table 4-4
TASK ANALYSIS (Continued)

SCHEDULED MAINTENANCE (SM)				
FUNCTION & /TASK/ SUB-TASK NUMBER	FREQUENCY	M/H'S - M/M'S	PERSONAL	ELAPSED TIME
ELECTRICAL POWER SYSTEM (SM) 2.4.2 (CONTINUED)				
. FUNCTIONAL (CONTINUED)				
. POWER-UP COMPLETE ELECTRICAL POWER SYSTEM AND VERIFY PROPER READOUT ON POWER SYSTEM MONITORS BY SWITCHING ALL BUSSES.		10 M/M'S	1	10 MIN.
. REMOVE AND INSTALL CHARGEDAAND CHECKED OUT SET OF BATTERIES (36 2ND STAGE - 4 1ST STAGE).		8 M/M'S	4	2 HRS.
. REPLACE THE FOLLOWING EQUIPMENT AT THE INDICATED FREQUENCIES: . APU GENERATOR . RELAY PANELS (FILTER) . POWER SYSTEM MONITORS	1000 HRS.	40 M/M'S	2	20 MIN.
. PERFORM POSITION, LANDING AND INTERIOR LIGHT OPERATIONAL CHECK.		20 M/M'S	2	10 MIN.

Table 4-4
TASK ANALYSIS (Continued)

SCHEDULED MAINTENANCE (SM)				
FUNCTION & /TASK/ SUB-TASK NUMBER	FREQUENCY	M/H'S - M/M'S	PERSONAL	ELAPSED TIME
GUIDANCE AND NAVIGATION AND CONTROL SYSTEM (SM) 2.4.3		10.5 M/H'S	-	4.4 HRS.
. VISUAL	AEF	4.3 M/H'S	-	1.3 HRS.
. INSP. DESICCANT ON HORIZON SENSOR FOR PROPER CONDITION		10 M/M'S	1	10 MIN.
. INSP. FOR PROPER HORIZON SENSOR ALIGNMENT IN RESPECT TO THE INERTIAL PLATFORM.		3 M/H'S	2	1.5 HRS.
. INSP. ALL EQUIPMENT FOR SECURITY OF MOUNTING INERTIAL MEASURING UNIT GUIDANCE AND NAVIGATION COMPUTER RENDEZVOUS RADAR IR HORIZON SENSORS STAR TRACKER AIR DATA SYSTEM TACAN/VORTAC NAVIGATION AIDS AIR TRAFFIC CONTROL TRANSPONDER RADIO ALTIMETER AILS INTEGRATED AUTOMATIC LANDING FLIGHT CONTROL ELECTRONICS RATE GYRO PACKAGE CONTROL DRIVERS HAND CONTROLLER RANGE RATE INDICATOR ATTITUDE DIRECTOR INDICATOR FLIGHT DIRECTOR CONTROLLER GUIDANCE COMPARATOR INDICATOR INCREMENTAL VELOCITY INDICATOR		30 M/M'S	2	15 MIN.
. INSP. ALL VISIBLE WIRE BUNDLES FOR CHAFING, LOOSE OR BROKEN TIES AND ANCHOR CLIPS AND LOOSE OR CORRODED CONNECTORS.		1 M/H	2	30 MIN.

Table 4-4
TASK ANALYSIS (Continued)

SCHEDULED MAINTENANCE (SM)

FUNCTION & /TASK/ SUB-TASK NUMBER	FREQUENCY	M/H'S - M/M'S	PERSONAL	ELAPSED TIME
GUIDANCE AND NAVIGATION AND CONTROL SYSTEM (SM) 2.4.3 (CONTINUED)				
. VISUAL (CONTINUED)				
. INSP. FOR SECURITY OF ALL CONNECTORS AND TERMINALS.	AEF	30 M/M'S	2	15 MIN.
. CHECK INERTIAL PLATFORM FOR PROPER PRESSURE.		05 M/M'S	1	5 MIN.
. CHECK RENDEZVOUS RADAR FOR PROPER PRESSURE.		05 M/M'S	1	5 MIN.
. INSP. ALL SWITCHES FOR SECURITY OF MOUNTING AND FUSE AND CIRCUIT BREAKERS FOR BLOWN OR UNLATCHED CONDITION.		30 M/M'S	2	15 MIN.
. INSP. PITOT AND STATIC (VISIBLE) LINES FOR DENTS, BENDS, CHAFING AND SECURE CONNECTIONS.		20 M/M'S	2	10 MIN.
. INSP. PITOT AND STATIC PRESSURE PORTS FOR SECURITY AND OBSTRUCTIONS.		10 M/M'S	2	5 MIN.
. INSP. ALL INDICATORS FOR SECURE KNOBS, SLIPPED OR MISSING RANGE MARKINGS AND CRACKED GLASS.		10 M/M'S	1	10 MIN.
. INSP. AIR DATA COMPUTER AIR TEST FITTINGS FOR SECURITY AND LOOSE OR MISSING DUST CAPS.		03 M/M'S	1	3 MIN.
. INSP. OPTICAL SIGHT FOR SECURITY OF MOUNTING AND EVIDENCE OF MISALIGNMENT.		05 M/M'S	1	5 MIN.
. INSP. OPTICAL SIGHT LENS FOR CLEANLINESS.		02 M/M'S	1	2 MIN.
. INSP. OPTICAL SIGHT ELECTRICAL CONNECTOR FOR SECURITY AND EVIDENCE OF CORROSION.		02 M/M'S	1	2 MIN.

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Table 4-4
TASK ANALYSIS (Continued)

SCHEDULED MAINTENANCE (SM)				
FUNCTION & /TASK/ SUB-TASK NUMBER	FREQUENCY	M/H'S - M/M'S	PERSONAL	ELAPSED TIME
GUIDANCE AND NAVIGATION AND CONTROL SYSTEM (SM) 2.4.3 (CONTINUED)				
. FUNCTIONAL	AEF			
. UTILIZING THE ONBOARD CHECKOUT SYSTEM, PERFORM A COMPLETE OPERATIONAL CHECK AND VERIFY PROPER READOUT, OPERATION AND RESPONSE.		4 M/H'S 4 M/H'S	2 2	2 HRS. 2 HRS.
. REPLACE THE FOLLOWING EQUIPMENT AT THE INDICATED FREQUENCIES:				
. INERTIAL MEASURING UNIT (IMU)				
. GUIDANCE AND NAVIGATION COMPUTER				
. RENDEZVOUS RADAR				
. IR HORIZON SENSORS				
. STAR TRACKER				
. AIR DATA SYSTEM				
. TACAN/VORTAC NAVIGATION AIDS				
. AIR TRAFFIC CONTROL TRANSPONDER				
. RADIO ALTIMETER				
. AILS INTEGRATED AUTOMATIC LANDING SYSTEM				
. FLIGHT CONTROL ELECTRONICS				
. RATE GYRO PACKAGE				
. CONTROL DRIVERS				
. HAND CONTROLLER				
. RANGE RATE INDICATOR				
. ATTITUDE DIRECTOR INDICATOR				
. FLIGHT DIRECTOR CONTROLLER				
. GUIDANCE COMPARATOR INDICATOR				
. INCREMENTAL VELOCITY INDICATOR				

Table 4-4
TASK ANALYSIS (Continued)

SCHEDULED MAINTENANCE (SM)				
FUNCTION & /TASK/ SUB-TASK NUMBER	FREQUENCY	M/H'S - M/M'S	PERSONAL	ELAPSED TIME
COMMUNICATION SYSTEM (SM) 2.4.4		4.2 M/H'S	-	2.3 HRS.
. VISUAL	AEF	1.2 M/H'S	-	53 MIN.
. INSP. ALL VISIBLE WIRE BUNDLES AND COAX CABLES FOR CHAFING, LOOSE OR BROKEN TIES AND ANCHOR CLIPS AND LOOSE OR CORRODED CONNECTORS.		1 M/H	2	30 MIN.
. INSP. ALL ANTENNAS FOR CRACKS, CLEANLINESS, AND SECURITY.		30 M/M'S	2	15 MIN.
. INSP. THE FOLLOWING EQUIPMENT FOR SECURITY OF ATTACHMENT:		1 M/H	2	30 MIN.
. VOICE AND DATA CONTROL CENTER				
. INTERCOM AND HEADSETS				
. UHF VOICE AND DATA TRANSCEIVERS				
. UHF MULTIPLEXER				
. UHF ANTENNAS				
. UHF ANTENNA SWITCHES				
. FLIGHT RECORDER				
. DIGITAL ENCODER/DECODER				
. SHF VOICE AND DATA TRANSCEIVERS				
. SHF DIPLEXER				
. SHF HIGH GAIN ANTENNA				
. PRINTER				
. INSP. ALL SWITCHES FOR SECURITY OF MOUNTING AND FUSE AND CIRCUIT BREAKERS FOR BLOWN OR UNLATCHED CONDITION.		30 M/M'S	2	15 MIN.
. INSP. ALL SELECTOR CONTROLS FOR SECURE KNOBS.		10 M/M'S	1	10 MIN.
. INSP. EXTERNAL INTERCOMMUNICATION RECEPTACLE FOR SECURITY AND CLEANLINESS: DUST CAP FOR SECURITY.		02 M/M'S	1	2 MIN.

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Table 4-4

TASK ANALYSIS (Continued)

SCHEDULED MAINTENANCE (SM)				
FUNCTION & /TASK/ SUB-TASK NUMBER	FREQUENCY	M/H'S - M/M'S	PERSONAL	ELAPSED TIME
COMMUNICATION SYSTEM (SM) 2.4.4 (CONTINUED)				
. FUNCTIONAL	AEF	1 M/H	-	30 MIN.
. PERFORM A COMPLETE OPERATIONAL CHECK AND VERIFY PROPER RESPONSE OF ALL COMMUNICATION SYSTEMS.		1 M/H	2	30 MIN.
. REPLACE THE FOLLOWING EQUIPMENT AT THE INDICATED FREQUENCIES:				
. VOICE AND DATA CONTROL CENTER				
. INTERCOM AND HEADSETS				
. UHF VOICE AND DATA TRANSCEIVERS				
. UHF MULTIPLEXER				
. UHF ANTENNAS				
. UHF ANTENNA SWITCHES				
. FLIGHT RECORDER				
. DIGITAL ENCODER/DECODER				
. SHF VOICE AND DATA TRANSCEIVERS				
. SHF DIPLEXER				
. SHF HIGH GAIN ANTENNA				
. PRINTER				

Table 4-4

TASK ANALYSIS (Continued)

SCHEDULED MAINTENANCE (SM)				
FUNCTION & /TASK/ SUB-TASK NUMBER	FREQUENCY	M/H'S - M/M'S	PERSONAL	ELAPSED TIME
ENVIRONMENTAL CONTROL AND LIFE SUPPORT SYSTEM (SM) 2.4.5		28.4 M/H'S	-	6.8 HRS.
. VISUAL	AEF	10.2 M/H'S	-	1.5 HRS.
. INSP. ALL VISIBLE WIRE BUNDLES FOR CHAFING, LOOSE OR BROKEN TIES AND ANCHOR CLIPS AND LOOSE OR CORRODED CONNECTORS.		4 M/H'S	8	30 MIN.
. INSP. FOR OBVIOUS DAMAGE.		1 M/H	4	15 MIN.
. INSP. ALL VISIBLE LINES AND HOSES FOR CHAFING, DETERIORATION AND FRAYING.		10 M/M'S	2	5 MIN.
. INSP. ALL SERVICE PORTS FOR OBVIOUS DAMAGE AND SECURITY OF DUST CAPS.		03 M/M'S	1	3 MIN.
. INSP. THE FOLLOWING EQUIPMENT FOR SECURITY OF ATTACHMENT:		4 M/H'S	8	30 MIN.
. GAS SUPPLY AND CONTROL				
. PRIMARY OXYGEN TANKS				
. PRESSURE SENSORS				
. OXYCEN AND NITROGEN VALUES				
. OXYGEN AND NITROGEN PRESSURE REGULATORS				
. SECONDARY OXYGEN TANKS				
. PRIMARY NITROGEN TANKS				
. GAS PROCESSING				
. COMPRESSORS				
. COMPRESSORS POWER SUPPLIES				
. CO_2 ABSORBER				
. SOLID TRAPS				
. CABIN FANS				
. FILTERS				
. HEAT TRANSPORT				
. COOLING PUMPS				
. COOLING PUMPS POWER SUPPLIES				
. COOLANT VALVES				

Table 4-4

TASK ANALYSIS (Continued)

SCHEDULED MAINTENANCE (SM)

FUNCTION & /TASK/ SUB-TASK NUMBER	FREQUENCY	M/H'S - M/M'S	PERSONAL	ELAPSED TIME
ENVIRONMENTAL CONTROL AND LIFE SUPPORT SYSTEM (SM) 2.4.5 (CONTINUED)				
. HEAT TRANSPORT (CONTINUED)	AEF			
. COOLANT FILTERS				
. WATER EVAPORATORS (2ND STAGE)				
. COLDPLATES				
. AMMONIA EVAPORATOR				
. THERMAL WALL PANELS				
. HEAT EXCHANGERS				
. PRESSURE AND TEMPERATURE SENSORS				
. ACCUMULATOR/RESERVOIR				
. RADIATORS (2ND STAGE)				
. WATER AND WASTE MANAGEMENT				
. PORTABLE WATER SUBSYSTEM				
. COOLING WATER SUPPLY SUBSYSTEM				
. HUMIDITY CONDENSATION SUBSYSTEM				
. URINE RECEIVER				
. VALVES (WASTE)				
. BELLOWS (WASTE)				
. STORAGE TANKS (WASTE)				
. INSP. FOR OBVIOUS LEAKS		1 M/H	2	30 MIN.
. INSP. ALL TEMP. SENSORS FOR PROPER BONDING		40 M/M'S	2	20 MIN.
. INSP. LOX SYSTEM FOR PROPER QUANTITY (PRIMARY)		02 M/M'S	1	2 MIN.
. INSP. GOX SYSTEM FOR PROPER PRESSURE (SECONDARY)		02 M/M'S	1	2 MIN.
. INSP. OXYGEN VENTS FOR OBSTRUCTIONS OR OBVIOUS DAMAGE		05 M/M'S	1	5 MIN.
. FUNCTIONAL		10.9 M/H'S	-	2.9 HRS.
. CLEAN OR REPLACE FILTERS (N_2, O_2, COOLANT AND H_2O)		4 M/H'S	8	30 MIN.
. REPLACE LIOH CARTRIDGE		3 M/H'S	2	1.5 HRS.

Table 4-4

TASK ANALYSIS (Continued)

SCHEDULED MAINTENANCE (SM)				
FUNCTION & /TASK/ SUB-TASK NUMBER	FREQUENCY	M/H'S - M/M'S	PERSONAL	ELAPSED TIME
ENVIRONMENTAL CONTROL AND LIFE SUPPORT SYSTEM (SM) 2.4.5 (CONTINUED)				
. FUNCTIONAL (CONTINUED)				
. REPLACE PO_2 SENSOR PRIOR TO EACH FLIGHT. UNIT MUST BE REPLACED IF EXPOSED TO ATMOSPHERE FOR MORE THAN 35 DAYS. AMPLIFIER AND SENSOR MUST BE REPLACED AS A SET DUE TO CALIBRATION.	AEF	1 M/H	1	1 HR.
. PERFORM A COOLANT SAMPLE TEST TO ANALYZE EVIDENCE OF BREAKDOWN OF PROPERTIES AND CONTAMINATION.		30 M/M'S	1	30 MIN.
. DESERVICE AND STERILIZE WATER MANAGEMENT SYSTEM.		2 M/H'S	2	1 HR.
. UTILIZING THE ONBOARD CHECKOUT, PERFORM AN OPERATIONAL TEST OF THE PRIMARY AND SECONDARY SYSTEMS.		10 M/M'S	1	10 MIN.
. AFTER CHECKOUT ASCERTAIN THAT THE LOX AND GOX IS SERVICED FOR FLIGHT. NOTE PLACE VENT VALVE IN FLIGHT POSITION AFTER SERVICING.		4 M/H'S	2	2 HRS.
. UTILIZING THE ONBOARD CHECKOUT SYSTEM, PERFORM AN OPERATIONAL TEST OF THE COOLANT SYSTEM AND VERIFY PROPER TEMP. AND FLOW.		10 M/M'S	1	10 MIN.
. SERVICE COOLANT SYSTEM AS REQUIRED.		6 M/H'S	2	3 HRS.
. PERFORM AN OPERATIONAL TEST OF THE WATER MANAGEMENT SYSTEM.		05 M/M'S	1	5 MIN.
. SERVICE WATER SYSTEM FOR FLIGHT NOTE REPLACE WATER PRIOR TO FLIGHT IF SERVICED FOR MORE THAN 14 DAYS.		30 M/M'S	2	15 MIN.

Table 4-4

TASK ANALYSIS (Continued)

SCHEDULED MAINTENANCE (SM)

FUNCTION & /TASK/ SUB-TASK NUMBER	FREQUENCY	M/H'S - M/M'S	PERSONAL	ELAPSED TIME
ENVIRONMENTAL CONTROL AND LIFE SUPPORT SYSTEM (SM)				
2.4.5 (CONTINUED)				
. FUNCTIONAL (CONTINUED)				
. REPLACE THE FOLLOWING EQUIPMENT AT THE INDICATED FREQUENCIES:	AEF			
. OXYGEN VALVES				
. NITROGEN VALVES				
. OXYGEN REGULATOR				
. NITROGEN REGULATORS				
. COMPRESSORS				
. COMPRESSOR POWER SUPPLIES				
. CABIN FANS				
. COOLING PUMPS				
. COOLING PUMP POWER SUPPLIES				
. COOLANT VALVES				
. PORTABLE WATER SUBSYSTEM				
. URINE VALVES				

Table 4-4

TASK ANALYSIS (Continued)

SCHEDULED MAINTENANCE (SM)				
FUNCTION & /TASK/ SUB-TASK NUMBER	FREQUENCY	M/H'S - M/M'S	PERSONAL	ELAPSED TIME
LANDING GEAR SYSTEM (SM) 2.4.6		21 M/H'S	-	8.9 HRS.
. VISUAL	AEF	3.5 M/H'S	-	1.4 HRS.
. INSP. UNLATCH MECHANISMS FOR CRACKS; LATCH ROLLER AND MECHANISM FOR CLEANLINESS AND BINDING.		40 M/M'S	4	10 MIN.
. INSP. LANDING GEAR EMERGENCY PNEUMATIC BOTTLE FOR NICKS, LOOSE OR CRACKED ATTACHMENT BANDS AND PROPERLY POSITIONED ANTI-CHAFE BANDS AND MOISTURE DRAINED; RELIEF VALVE FOR SECURITY, CRACKS, CORROSION AND LEAKAGE.		05 M/M'S	1	5 MIN.
. INSP. LANDING GEAR DOORS FOR SECURITY, CRACKS, CORROSION AND LOOSE OR MISSING RIVETS. LINK RODS FOR DISTORTION AND LOOSE ROD ENDS.		08 M/M'S	4	2 MIN.
. INSP. STRUT DOORS FOR SECURITY, CRACKS, CORROSION AND LOOSE OR MISSING RIVETS - STRUT DOOR GUIDE BLOCKS FOR WEAR, BINDING, CORROSION AND DISTORTION.		08 M/M'S	4	2 MIN.
. INSP. ACTUATOR ASSEMBLIES FOR LEAKAGE, END FITTINGS (DOWN LOCK) FOR SECURITY, LOWER AND UPPER ATTACH POINTS FOR DISTORTION, CRACKS, BEARING SEIZURE, SHUTTLE VALVE FOR LEAKAGE.		12 M/M'S	4	3 MIN.
. INSP. ACTUATOR LIMIT SWITCHES AND CAM MECHANISMS FOR CORROSION AND SECURITY.		08 M/M'S	4	2 MIN.
. INSP. ACTUATOR CONNECTING HYDRAULIC AND PNEUMATIC LINES FOR LEAKAGE AND SECURITY. ELECT. CONNECTORS FOR SECURITY AND HARNESS FOR DETERIORATION.		08 M/M'S	4	2 MIN.
. INSP. UPLATCH MECHANISM CYLINDER AND SHUTTLE VALVE FOR LEAKAGE AND SECURITY SEQUENCE VALVE FOR LEAKAGE AND SECURITY.		12 M/M'S	4	3 MIN.

Table 4-4

TASK ANALYSIS (Continued)

SCHEDULED MAINTENANCE (SM)				
FUNCTION & /TASK/ SUB-TASK NUMBER	FREQUENCY	M/H'S - M/M'S	PERSONAL	ELAPSED TIME
LANDING GEAR SYSTEM (SM) 2.4.6 (CONTINUED)				
. VISUAL (CONTINUED)				
. INSP. UPLATCH UP-LIMIT SWITCH AND CONNECTORS FOR SECURITY AND CORROSION. ELECT. HARNESS FOR DETERIORATION.	AEF	08 M/M'S	4	2 MIN.
. INSP. MLC FOR CORROSION, NICKED OR SCORED PISTON, CRACKS, LEAKS AND EVIDENCE OF PISTON BOTTOMING.		09 M/M'S	3	3 MIN.
. INSP. MLG STRUT UPPER ATTACHMENT TRUNNIONS AND DRAG BRACE FOR DISTORTION.		12 M/M'S	3	4 MIN.
. INSP. TORQUE LINKS AND PINS FOR DISTORTION, LEAKAGE, CRACKS AND SECURITY. SAFETY SWITCH FOR SECURITY AND ELECT. HARNESS FOR DETERIORATION.		09 M/M'S	3	3 MIN.
. INSP. MLG SHRINK LINKS, PINS, STUDS AND ROD END FITTINGS FOR CRACKS, DISTORTION AND SECURITY.		09 M/M'S	3	3 MIN.
. INSP. ANTI-SKID SENSOR ELECTRICAL HARNESS FOR DETERIORATION AND SECURITY.		06 M/M'S	3	2 MIN.
. INSP. LANDING GEAR EMERGENCY PNEUMATIC BOTTLE RELIEF VALVE FOR SECURITY, CRACKS, CORROSION AND LEAKAGE.		02 M/M'S	1	2 MIN.
. INSP. BRAKE POWER DUAL CONTROL VALVES FOR LEAKAGE. LINKAGE FOR DISTORTION AND SECURITY.		06 M/M'S	3	2 MIN.
. INSP. ANTI-SKID CONTROL VALVE FOR LEAKAGE AND SECURITY. ELECTRICAL CONNECTOR FOR SECURITY.		06 M/M'S	3	2 MIN.

Table 4-4

TASK ANALYSIS (Continued)

SCHEDULED MAINTENANCE (SM)				
FUNCTION & /TASK/ SUB-TASK NUMBER	FREQUENCY	M/H'S - M/M'S	PERSONAL	ELAPSED TIME
LANDING GEAR SYSTEM (SM) 2.4.6 (CONTINUED)				
. VISUAL (CONTINUED)				
. INSP. NOSE GEAR UPLIMIT SWITCH FOR CORROSION HARNESS FOR DETERIORATION AND SECURITY.	AEF	01 M/M'S	1	1 MIN.
. INSP. NOSE GEAR UPLATCH CYLINDER AND SHUTTLE VALVE FOR LEAKAGE AND SECURITY. MECHANISM LINKAGE FOR CORROSION, CRACKS, DISTORTION AND SECURITY; ATTACHMENT POINT HOLES FOR ELONGATION.		02 M/M'S	1	2 MIN.
. INSP. NOSE LANDING GEAR STRUT FOR CORROSION.		03 M/H'S	1	3 MIN.
. INSP. NOSE LANDING GEAR STRUT UPPER TRUNNIONS ATTACHING PINS AND LOCK BOLTS FOR DISTORTION AND SECURITY. ACTUATOR ATTACHMENT PIN AND BOSS FOR CRACKS AND LOOSENESS.		03 M/M'S	1	3 MIN.
. INSP. STEERING LINKAGE FOR SCORING AND PITTING.		10 M/M'S	1	10 MIN.
. INSP. NOSE GEAR STRUT FOR LEAKAGE; POLISHED SURFACE OR STRUT AND DRAG BRACE ACTUATOR CLEANED WITH CLOTH MOISTENED WITH (ORONITE 70 U).		01 M/M	1	1 MIN.
. NOSE GEAR STRUT FOR NORMAL EXTENSION.		01 M/M	1	1 MIN.
. NOSE GEAR TIRES FOR CUTS AND IMBEDDED OBJECTS.		03 M/M'S	1	3 MIN.
. NOSE GEAR TIRES FOR SPECIFIED PRESSURE; VALVE STEM CAPS FOR SECURITY.		02 M/M'S	1	2 MIN.
. EMERGENCY BRAKE ACCUMULATOR FOR SPECIFIED PRESSURE.		01 M/M	1	1 MIN.
. ENSURE EMERGENCY BRAKE VALVE IS IN PROPER POSITION.		01 M/M	1	1 MIN.
NOTE: 1. PRIOR TO LAUNCH (PTL)				

Table 4-4

TASK ANALYSIS (Continued)

SCHEDULED MAINTENANCE (SM)				
FUNCTION & /TASK/ SUB-TASK NUMBER	FREQUENCY	M/H'S - M/M'S	PERSONAL	ELAPSED TIME
LANDING GEAR SYSTEM (SM) 2.4.6 (CONTINUED)				
. VISUAL (CONTINUED)				
. INSP. LANDING GEAR EMERGENCY SYSTEM PNEUMATIC GAGE FOR SPECIFIED INDICATION.	AEF	01 M/M	1	1 MIN.
. INSP. MLG TIRES FOR CUTS AND IMBEDDED OBJECTS.		16 M/M'S	3	2 MIN.
. INSP. MLG TIRES FOR SPECIFIED PRESSURE; VALVE STEM CAPS FOR SECURITY.		45 M/M'S	3	15 MIN.
. INSP. MLG STRUTS FOR NORMAL EXTENSION.		09 M/M'S	3	3 MIN.
. INSP. MLG STRUTS FOR LEAKAGE; POLISHED SURFACE OF STRUT AND SIDE BRACE ACTUATOR CLEANED WITH CLOTH MOISTENED WITH (ORONITE 70 U).		06 M/M'S	3	2 MIN.
. NOSE GEAR TIRES FOR CUTS, WEAR OR MISSING TREAD; VALVE STEM CAPS FOR SECURITY.		01 M/M	1	1 MIN.
. INSP. NOSE GEAR STRUT FOR NORMAL EXTENSION.		01 M/M	1	1 MIN.
. INSP. NOSE GEAR STRUT, DRAG BRACE ACTUATOR AND UPLOCK CYLINDER FOR LEAKAGE.		03 M/M'S	1	3 MIN.
. INSP. NOSE GEAR STEERING POWER UNIT, COMPENSATOR AND SELECTOR VALVE FOR LEAKAGE AND SECURITY. FOLLOW POTENTIOMETER AND ELECTRICAL CONNECTOR FOR SECURITY AND CORROSION.		04 M/M'S	1	4 MIN.
. INSP. NOSE GEAR AND ACTUATORS POLISHED SURFACES CLEANED WITH CLOTH MOISTENED WITH (ORONITE 70 U).		02 M/M'S	1	2 MIN.
. INSP. EMERGENCY BRAKE ACCUMULATOR GAGE FOR SPECIFIED PRESSURE WITH HYDRAULIC SYSTEM PRESSURE DEPLETED.		01 M/M	1	1 MIN.

Table 4-4

TASK ANALYSIS (Continued)

SCHEDULED MAINTENANCE (SM)				
FUNCTION & /TASK/ SUB-TASK NUMBER	FREQUENCY	M/H'S - M/M'S	PERSONAL	ELAPSED TIME
LANDING GEAR SYSTEM (SM) 2.4.6 (CONTINUED)				
. VISION (CONTINUED)				
. INSP. NOSE GEAR DOORS FOR CRACKS, DENTS, SECURITY AND HOT SPOTS.	AEF	02 M/M'S	1	2 MIN.
. INSP. MAIN TIRES FOR CUTS, WEAR OR MISSING TREAD; VALVE STEM CAPS FOR SECURITY.		16 M/M'S	4	4 MIN.
. INSP. MAIN STRUTS FOR NORMAL EXTENSION.		08 M/M'S	4	2 MIN.
. INSP. ACCESSIBLE BRAKE COMPONENTS FOR LEAKAGE AND EVIDENCE OF HEAT DAMAGE; ANTI-SKID SENSORS AND ELECTRICAL CONNECTORS FOR SECURITY AND DISTORTION.		20 M/M'S	2	10 MIN.
. INSP. MLG DOORS FOR CRACKS, DENTS, SECURITY AND HOT SPOTS.		09 M/M'S	3	3 MIN.
. INSP. MLG STRUTS, SIDE BRACES, ACTUATOR, AND UPLOCK CYLINDER FOR LEAKAGE; CLEAN POLISHED SURFACES WITH CLOTH MOISTENED WITH (ORONITE 70 U).		15 M/M'S	3	5 MIN.
. FUNCTIONAL		12.5 M/H'S	-	4.7 HRS.
. REMOVE MLG WHEEL ASSEMBLIES AND INSP. BRAKE ASSEMBLIES (ROTORS & STATORS) FOR WARPAGE, CRACKS, DISTORTION, LEAKS, WEAR AND ANY OBVIOUS DAMAGE. ANY INDICATIONS OF THE ABOVE REMOVE AND REPLACE BRAKE ASSEMBLIES.		12 M/H'S	3	4 HRS.

Table 4–4

TASK ANALYSIS (Continued)

SCHEDULED MAINTENANCE (SM)

FUNCTION & /TASK/ SUB-TASK NUMBER	FREQUENCY	M/H'S - M/M'S	PERSONAL	ELAPSED TIME
LANDING GEAR SYSTEM (SM) 2.4.6 (CONTINUED)				
. FUNCTIONAL (CONTINUED)				
. *PERFORM LANDING GEAR FUNCTIONAL CHECK (NORMAL AND EMERGENCY) VERIFY FULL TRAVEL, UP-LOCK AND DOWN-LOCK.	AEF	2.5 M/H'S	5	30 MIN.
. CHECK LANDING GEAR DOORS FOR CHAFING AND FLUSH FIT. VERIFY PROPER CLEARANCE IN THE THERMAL EXPANSION AREAS.				
. *PERFORM NOSE GEAR STEERING OPERATION AND VERIFY PROPER TRAVEL IN THE FULL LEFT AND FULL RIGHT POSITIONS. VERIFY NOSE GEAR RETURNS TO NEUTRAL POSITION.		10 M/M'S	2	5 MIN.
. CLEAN OR REPLACE NOSE GEAR STEERING FILTER ELEMENT		15 M/M'S	1	15 MIN.
. PERFORM BRAKE OPERATIONAL CHECK AND VERIFY NO LEAKAGE AND PROPER ANTI-SKID OPERATION.		30 M/M'S	2	15 MIN.
NOTE: * PERFORM OPERATIONAL CHECK VIA CHECKOUT AND MANUAL.				

Table 4–4

TASK ANALYSIS (Continued)

SCHEDULED MAINTENANCE (SM)				
FUNCTION & /TASK/ SUB-TASK NUMBER	FREQUENCY	M/H'S - M/M'S	PERSONAL	ELAPSED TIME
AERODYNAMIC FLIGHT CONTROL SYSTEM (SM) 2.4.7		18.9 M/H'S	-	5.5 HRS.
. VISUAL	AEF	9.5 M/H'S	-	1.5 HRS.
. INSP. WING HINGE BOLTS, NUTS AND LOCK PINS FOR SECURITY AND OBVIOUS DAMAGE.		4 M'H'S	4	1 HR.
. INSP. WING HINGES AND ATTACH FITTINGS FOR CRACKS AND CORROSION: BONDING JUMPERS FOR WEAR AND SECURITY.		2 M/H'S	4	30 MIN.
. INSP. COMPLETE WING ASSEMBLIES FOR OBVIOUS DAMAGE.		1 M'H	2	30 MIN.
. INSP. WING DRIVE SYSTEM AND ACCESSIBLE LINES, HOSES AND ATTACHED HYDRAULIC FITTINGS FOR CUTS, CHAFING, FRAYING, LEAKAGE AND SECURITY.		2 M/H'S	4	30 MIN.
. INSP. WING SWIVEL FITTINGS FOR LEAKS AND SECURITY.		20 M/M'S	2	10 MIN.
. INSP. WING SERVO VALVES FOR LEAKAGE AND SECURITY.		20 M/M'S	2	10 MIN.
. INSP. AILERON HINGE BOLTS, NUTS AND LOCK PINS FOR SECURITY AND OBVIOUS DAMAGE.		30 M/M'S	2	15 MIN.
. INSP. AILERON HINGES AND ATTACH FITTINGS FOR CRACKS AND CORROSION; BONDING JUMPERS FOR WEAR AND SECURITY.		30 M/M'S	2	15 MIN.
. INSP. COMPLETE AILERON SURFACES FOR DENTS, PITS & OBVIOUS DAMAGE.		10 M/M'S	2	5 MIN.
. INSP. AILERON POWER CYLINDERS AND ACCESSIBLE LINES, HOSES AND ATTACHED HYDRAULIC FITTINGS FOR CUTS, CHAFING, FRAYING, LEAKAGE AND SECURITY: ROD ENDS AND ATTACH FITTINGS FOR CRACKS.		30 M/M'S	2	15 MIN.
. INSP. AILERON SWIVEL FITTINGS FOR LEAKS AND SECURITY.		10 M/M'S	2	5 MIN

Table 4-4
TASK ANALYSIS (Continued)

SCHEDULED MAINTENANCE (SM)				
FUNCTION & /TASK/ SUB-TASK NUMBER	FREQUENCY	M/H'S - M/M'S	PERSONAL	ELAPSED TIME
AERODYNAMIC FLIGHT CONTROL SYSTEM (SM) 2.4.7 (CONTINUED)				
. VISUAL (CONTINUED)				
. INSP. AILERON SERVO VALVES FOR LEAKAGE AND SECURITY.	AEF	10 M/M'S	2	5 MIN.
. INSP. FLAP HINGE BOLTS, NUTS AND LOCK PINS FOR SECURITY AND OBVIOUS DAMAGE.		10 M/M'S	2	5 MIN.
. INSP. FLAP HINGES AND ATTACH FITTINGS FOR CRACKS AND CORROSION: BONDING JUMPERS FOR WEAR AND SECURITY.		20 M/M'S	2	10 MIN.
. INSP. COMPLETE FLAP SURFACES FOR DENTS, PITS AND OBVIOUS DAMAGE.		10 M/M'S	2	5 MIN.
. INSP. FLAP POWER CYLINDERS AND ACCESSIBLE LINES, HOSES, AND ATTACHED HYDRAULIC FITTINGS FOR CUTS, CHAFING, FRAYING AND SECURITY: ROD ENDS & ATTACH FITTINGS FOR CRACKS.		30 M/M'S	2	15 MIN.
. INSP. FLAP SWIVEL FITTINGS FOR LEAKS AND SECURITY.		10 M/M'S	2	5 MIN.
. INSP. EMERGENCY FLAP PNEUMATIC BOTTLE FOR NICKS AND LOOSE OR CRACKED ATTACHMENT BANDS AND PROPERLY POSITIONED ANTI-CHAFE STRIPS: MOISTURE DRAINED.		10 M/M'S	1	10 MIN.
. INSP. LEVEL OF HYDRAULIC FLUID.		01 M/M	1	1 MIN.
. INSP. PNEUMATIC BOTTLE PRESSURE.		01 M/M	1	1 MIN.
. INSP. ALL VISIBLE WIRE BUNDLES FOR CHAFING, LOOSE OR BROKEN TIES AND ANCHOR CLIPS AND LOOSE OR CORRODED CONNECTORS IN THE WING, FLAP & AILERON SYSTEMS.		1 M/H	6	10 MIN.
. INSP. ALL VISIBLE WIRE BUNDLES FOR CHAFING, LOOSE OR BROKEN TIES AND ANCHOR CLIPS AND LOOSE OR CORRODED CONNECTORS IN THE ELEVON SYSTEM.		30 M/M'S	2	15 MIN.

Table 4-4

TASK ANALYSIS (Continued)

SCHEDULED MAINTENANCE (SM)				
FUNCTION & /TASK/ SUB-TASK NUMBER	FREQUENCY	M/H'S - M/M'S	PERSONAL	ELAPSED TIME
AERODYNAMIC FLIGHT CONTROL SYSTEM (SM) 2.4.7 (CONTINUED)				
. VISUAL (CONTINUED)				
. INSP. ELEVON HINGE BOLTS, NUTS AND LOCK PINS FOR SECURITY AND OBVIOUS DAMAGE.	AEF	10 M/M'S	2	5 MIN.
. INSP. ELEVON HINGE AND ATTACH FITTINGS FOR CRACKS AND CORROSION: BONDING JUMPERS FOR WEAR AND SECURITY.		20 M/M'S	2	10 MIN.
. INSP. ELEVON DOR DENTS, PITS AND OBVIOUS DAMAGE.		10 M/M'S	2	5 MIN.
. INSP. ELEVON POWER CYLINDER AND ACCESSIBLE LINES, HOSES AND ATTACHED HYDRAULIC FITTINGS FOR CRACKS AND DISTORTION.		30 M/M'S	2	15 MIN.
. INSP. ELEVON SWIVEL FITTINGS FOR LEAKS AND SECURITY.		10 M/M'S	2	5 MIN.
. INSP. ELEVON SERVO VALVES FOR LEAKAGE AND SECURITY.		10 M/M'S	2	5 MIN.
. INSP. ELEVON FOR CORROSION AND LOOSE OR MISSING RIVETS OR SCREWS.		10 M/M'S	2	5 MIN.
. INSP. ELEVON TRIM ACTUATOR ELECTRICAL CONNECTOR FOR SECURITY.		05 M/M'S	1	5 MIN.
. INSP. ALL VISIBLE WIRE BUNDLES FOR CHAFING, LOOSE OR BROKEN TIES AND ANCHOR CLIPS AND LOOSE OR CORRODED CONNECTORS ON THE RUDDER SYSTEM.		30 M/M'S	2	15 MIN.
. INSP. AERO SURFACE HINGE BOLTS, NUTS AND LOCK PINS FOR SECURITY AND OBVIOUS DAMAGE.		10 M/M'S	2	5 MIN.
. INSP. AERO SURFACE HINGE AND ATTACH FITTINGS FOR CRACKS AND CORROSION: BONDING JUMPERS FOR WEAR AND SECURITY.		20 M/M'S	2	10 MIN.

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Table 4-4

TASK ANALYSIS (Continued)

SCHEDULED MAINTENANCE (SM)				
FUNCTION & /TASK/ SUB-TASK NUMBER	FREQUENCY	M/H'S - M/M'S	PERSONAL	ELAPSED TIME
AERODYNAMIC FLIGHT CONTROL SYSTEM (SM) 2.4.7 (CONTINUED)				
. VISUAL (CONTINUED)				
. INSP. AERO SURFACE FOR DENTS, PITS AND OBVIOUS DAMAGE.	AEF	10 M/M'S	2	5 MIN.
. INSP. AERO SURFACE POWER CYLINDER AND ACCESSIBLE LINES AND ATTACHED HYDRAULIC FITTINGS FOR CUTS, CHAFING, FRAYING, LEAKAGE AND SECURITY.		30 M/M'S	2	15 MIN.
. ROD ENDS AND ATTACH FITTINGS FOR CRACKS AND DISTORTION.		20 M/M'S	2	10 MIN.
. INSP. AERO SURFACE SWIVEL FITTINGS FOR LEAKS AND SECURITY.		05 M/M'S	1	5 MIN.
. INSP. AERO SURFACE SERVO VALVES FOR LEAKS AND SECURITY.		05 M/M'S	1	5 MIN.
. INSP. AERO SURFACE FOR CORROSION AND LOOSE OR MISSING RIVETS OR SCREWS.		05 M/M'S	1	5 MIN.
. INSP. AERO SURFACE DAMPER FOR LEAKAGE AND ROD ENDS FOR SECURITY.		05 M/M's	1	5 MIN.
. INSP. AERO SURFACE FEEL CYLINDER FOR LEAKAGE AND SECURITY.		05 M/M'S	1	5 MIN.
. FUNCTIONAL		1.08	-	35 MIN.
. PERFORM A COMPLETE WING OPERATIONAL AND VERIFY PROPER WING TRAVEL (EXTENDED AND RETRACTED), SEQUENCE AND NO LEAKS.		30 M/M'S	3	10 MIN.
. PERFORM A COMPLETE AILERON OPERATIONAL AND VERIFY PROPER AILERON TRAVEL (FULL UP, FULL DOWN & NEUTRAL) AND NO LEAKS.		15 M/M'S	3	5 MIN.
. INSP. CONTROL STICKS FOR LOOSENESS AT SOCKET AND CLEANLINESS.		02 M/M'S	2	1 MIN.
. PERFORM AILERON TRIM CHECK.		2 M/M'S	1	2 MIN.

Table 4-4

TASK ANALYSIS (Continued)

SCHEDULED MAINTENANCE (SM)				
FUNCTION & /TASK/ SUB-TASK NUMBER	FREQUENCY	M/H'S - M/M'S	PERSONAL	ELAPSED TIME
AERODYNAMIC FLIGHT CONTROL SYSTEM (SM) 2.4.7 (CONTINUED)				
. FUNCTIONAL (CONTINUED)				
. PERFORM A COMPLETE FLAP OPERATIONAL AND VERIFY PROPER FLAP TRAVEL (FULL UP, FULL DOWN & % OF FLAP VIA INDICATOR) & NO LEAKS.	AEF	20 M/M'S	4	5 MIN.
. PERFORM A COMPLETE ELEVON OPERATIONAL AND VERIFY PROPER TRAVEL (FULL UP, FULL DOWN & NEUTRAL) AND FREEDOM OF MOVEMENT.		20 M/M'S	2	10 MIN.
. PERFORM ELEVON TRIM OPERATION AND VERIFY NO CREEP.		02 M/M'S	1	2 MIN.
. PERFORM A COMPLETE AERO SURFACE OPERATIONAL AND VERIFY PROPER TRAVEL (FULL LEFT, FULL RIGHT & NEUTRAL) AND FREEDOM OF MOVEMENT.		15 M/M'S	3	5 MIN.
. PERFORM AERO SURFACE PEDAL ADJUSTMENT AND VERIFY NO BINDING.		02 M/M'S	1	2 MIN.
. PERFORM AERO SURFACE TRIM OPERATION AND VERIFY PROPER OPERATION.		02 M/M'S	1	2 MIN.
NOTE THE ABOVE FUNCTIONAL CHECKS UTILIZE THE FLY-BY-WIRE & MANUAL OPERATIONAL WITH PRIMARY & SECONDARY HYDRAULIC SYSTEMS.				

Table 4-4

TASK ANALYSIS (Continued)

SCHEDULED MAINTENANCE (SM)				
FUNCTION & /TASK/ SUB-TASK NUMBER	FREQUENCY	M/H'S - M/M'S	PERSONAL	ELAPSED TIME
PROPULSION FLIGHT CONTROL SYSTEM (SM) 2.4.8		11.7 M/H'S	-	3.1 HRS.
. VISUAL	AEF	4.7 M/H'S	-	1.1 HRS.
. INSP. ALL VISIBLE WIRE BUNDLES FOR CHAFING, LOOSE OR BROKEN TIES AND ANCHOR CLIPS AND LOOSE OR CORRODED CONNECTORS.		1 M/H	2	30 MIN.
. INSP. FOR OBVIOUS DAMAGE.		2.2 M'H'S	9	15 MIN.
. INSP. ALL VISIBLE LINES FOR CHAFING AND DETERIORATION OR ANY OBVIOUS DAMAGE.		2 M/H'S	4	30 MIN.
. INSP. ALL SERVICE PORTS FOR OBVIOUS DAMAGE AND SECURITY OF DUST CAPS.		30 M/M'S	2	15 MIN.
. INSP. THE FOLLOWING EQUIPMENT FOR SECURITY OF ATTACHMENT: . LH_2 TANKS . LH_2 TANKS . TURBO COMPRESSOR . GAS GENERATOR . GH_2 & CO_2 HIGH PRESSURE ACCUMULATORS . REGULATORS . ENGINES		4 M/H'S	4	1 HR.
. INSP. FOR ANY EVIDENCE OF LEAKAGE		2 M/H'S	4	30 MIN.
. FUNCTIONAL		4.5 M/H'S	-	1.1 HRS.
. UTILIZING THE ONBOARD CHECKOUT, VERIFY PROPER IGNITER FIRING & SEQUENCING.		02 M/M'S	1	2 MIN.
. PERFORM A COMPLETE OPERATIONAL CHECK AND VERIFY PROPER READOUTS AND OPERATIONS.		30 M/M'S	3	10 MIN.
. BACK FLOW CLEAN OR REPLACE PROPULSION FILTERS.		4 M/H'S	4	1 HR.

Table 4-4

TASK ANALYSIS (Continued)

SCHEDULED MAINTENANCE (SM)				
FUNCTION & /TASK/ SUB-TASK NUMBER	FREQUENCY	M/H'S - M/M'S	PERSONAL	ELAPSED TIME
PROPULSION FUEL SYSTEM (SM) 2.4.9		16.25 M/H'S	-	3.25 HRS.
. VISUAL	AEF	10 HRS.	-	2.0 HRS.
. INSP. ALL VISIBLE WIRE BUNDLES FOR CHAFING, LOOSE OR BROKEN TIES AND ANCHOR CLIPS AND LOOSE OR CORRODED CONNECTORS.		6 M/H'S	6	1 HR.
. INSP. THE LIQUID, OXYGEN, AND HYDROGEN TANKS FOR DENTS, CRACKS, LEAKS AND ANY DETERIORATION AND DAMAGE.		9 M/H'S	6	1.5 HRS.
. INSP. ALL VISIBLE LINES FOR CHAFING AND DETERIORATION OR ANY OBVIOUS DAMAGE.		6 M/H'S	6	1 HR.
. INSP. ALL SERVICE PORTS FOR OBVIOUS DAMAGE AND SECURITY OF DUST CAPS.		1 M/H	2	30 MIN.
. INSP. THE HEAT EXCHANGERS FOR SECURITY LEAKS AND OBVIOUS DAMAGE.		15 M/M'S	1	15 MIN.
. FUNCTIONAL		30 M/M'S	-	.1 HRS.
. UTILIZING THE ONBOARD CHECKOUT SYSTEM, PERFORM A COMPLETE OPERATIONAL CHECK AND VERIFY PROPER READOUTS AND OPERATIONS. ASCERTAIN THAT THE PNEUMATIC CONTROL OPERATES PROPERLY AND IS SERVICED WITH HELIUM TO L%)) PSI.		30 M/M'S	5	6 MIN.

Table 4-4
TASK ANALYSIS (Continued)

SCHEDULED MAINTENANCE (SM)				
FUNCTION & /TASK/ SUB-TASK NUMBER	FREQUENCY	M/H'S - M/M'S	PERSONAL	ELAPSED TIME
MAIN PROPULSION SYSTEM (SM) 2.4.10		18.4 M/H'S	-	3.9 HRS.
. VISUAL	PTL	14.3 M/H'S	-	1.8 HRS.
. INSP. AUGMENTED SPARK IGNITION (ASI) ASSY.		2.0 M/H'S	8	15 MIN.
. INSP. THRUST CHAMBER ASSY.		1.3 M/H'S	8	10 MIN.
. INSP. PROPELLANT FEED SYSTEM INSTL.		2.0 M/H'S	8	15 MIN.
. INSP. ALL HOSES FOR FRAYING AND DETERIORATION.		1.3 M/H'S	8	10 MIN.
. INSP. ALL VALVES FOR OBVIOUS DAMAGE AND LEAKS.		1.3 M/H'S	8	10 MIN.
. INSP. FUEL AND OXIDIZER TURBOPUMP ASSEMBLIES.		1.3 M/H'S	8	10 MIN.
. INSP. START SYSTEM INSTALLATION.		1.3 M/H'S	8	10 MIN.
. INSP. CONTROL SYSTEM INSTALLATION.		1.3 M/H'S	8	10 MIN.
. INSP. INSTRUMENTATION INSTALLATION.		1.3 M/H'S	8	10 MIN.
. INSP. THE FUEL AND OXIDIZER SYSTEMS FOR OBVIOUS LEAKS.		1.3 M/H'S	8	10 MIN.
. FUNCTIONAL		4.1 M/H'S	-	2.1 HRS.
. PERFORM A FUNCTIONAL CHECK OF THE SPARK IGNITERS.		10 M/M'S	2	5 MIN.
. PERFORM A FUNCTIONAL CHEKC OF THE FUEL AND OXIDIZER FLOW TRANSDUCERS.		1.0 M/H	2	30 MIN.
. PERFORM A FUNCTIONAL CHECK OF THE FUEL AND OXIDIZER VALVE ASSEMBLIES.		10 M/M'S	2	5 MIN.
. PERFORM A FUNCTIONAL CHECK OF THE FUEL AND OXIDIZER TURBOPUMP ASSEMBLIES		20 M/M'S	2	10 MIN.
. PERFORM A FUNCTIONAL CHECK OF THE START SYSTEM.		10 M/M'S	1	10 MIN.
. PERFORM A FUNCTIONAL CHECK OF THE CONTROL SYSTEM.		1 M/H	2	30 MIN.
. PERFORM A FUNCTIONAL CHECK OF THE INSTRUMENTATION SYSTEM.		1 M/H	2	30 MIN.

Table 4-4

TASK ANALYSIS (Continued)

SCHEDULED MAINTENANCE (SM)				
FUNCTION & /TASK/ SUB-TASK NUMBER	FREQUENCY	M/H'S - M/M'S	PERSONAL	ELAPSED TIME
MAIN PROPULSION SYSTEM (SM) 2.4.10 (CONTINUED) . FUNCTIONAL (CONTINUED) . PERFORM NOZZLE EXTENSION CHECK. NOTE THE ABOVE FUNCTIONAL CHECKS ARE PERFORMED UTILIZING THE ONBOARD CHECKOUT.	PTL	20 M/M'S	2	10 MIN.

Table 4-4
TASK ANALYSIS (Continued)

SCHEDULED MAINTENANCE (SM)				
FUNCTION & /TASK/ SUB-TASK NUMBER	FREQUENCY	M/H'S - M/M'S	PERSONAL	ELAPSED TIME
ASSIST ENGINES FUEL SYSTEM (SM) 2.4.11		6.3 M/H'S	-	2.7 HRS.
. VISUAL	AEF	2.0 M/H'S	-	33 MIN.
. INSP. THE EXTERIOR OF THE FUSELAGE FOR FUEL LEAKAGE.		10 M/M'S	2	5 MIN.
. INSP. THE FUEL TANK DRAINS FOR LEAKAGE AND OBSTRUCTION.		05 M/M'S	1	5 MIN.
. INSP. MOTOR OPERATED FUEL SHUTOFF VALVES FOR EVIDENCE OF LEAKAGE; ELECTRICAL CONNECTORS FOR SECUIRTY.		1.3 M/M'S	8	10 MIN.
. INSP. FUEL MANIFOLD AND COUPLING CLAMPS FOR CRACKS, LEAKAGE, BROKEN OR MISSING SAFETY WIRE.		2 M/H'S	2	1 HR.
. INSP. FUEL LINES AND ELBOWS FOR LEAKAGE, FRAYED BONDING JUMPERS OR BROKEN SAFETY WIRE.		45 M/M'S	2	22.5 MIN.
. INSP. FUEL VENT LINES FOR EVIDENCE OF HEAT DAMAGE, CRACKS, CHAFING AND LEAKAGE.		10 M/M'S	1	10 MIN.
. INSP. ALL FUEL TANK AND MANIFOLD DRAIN LINE SHUTOFF VALVES SAFETIED.		20 M/M'S	2	10 MIN.
. INSP. FUEL PRESSURIZATION LINE FOR CRACKS.		10 M/M'S	1	10 MIN.
. INSP. FUEL DISCONNECT FITTING FOR CLEANLINESS, SCRATCHED OR NICKED SEAL SURFACE AND CRACKED FLANGE.		05 M/M'S	1	5 MIN.
. FUNCTIONAL	AEF & PTL	.9 M/H'S	-	15 MIN.
. VERIFY THE FUEL QUANTITY GAGES FOR PROPER INDICATION.	PTL	10 M/M'S	1	10 MIN.
. INSP. FUEL TANK DRAINS FOR CONTAMINATION.		20 M/M'S	2	10 MIN.

Table 4-4
TASK ANALYSIS (Continued)

SCHEDULED MAINTENANCE (SM)				
FUNCTION & /TASK/ SUB-TASK NUMBER	FREQUENCY	M/H'S - M/M'S	PERSONAL	ELAPSED TIME
ASSIST ENGINES FUEL SYSTEM (SM) 2.4.11 (CONTINUED)				
. FUNCTIONAL (CONTINUED)				
. VERIFY FUEL QUANTITY INDICATING SYSTEM TO ASSURE COMPLETE SERVICING BY ADDING INDICATED QUANTITY OF FUEL REMAINING IN TANKS BEFORE SERVICING AND THE AMOUNT SERVICED.	AEF	10 M/M'S	1	10 MIN.
. VERIFY ELECTRICAL DRIVEN FUEL TRANSFER PUMPS FOR PROPER LIGHT INDICATION.	PTL	05 M/M'S	1	5 MIN.
. VERIFY PROPER OPERATION OF FUEL CONTROL VALVES.	AEF	10 M/M'S	2	5 MIN.

Table 4-4

TASK ANALYSIS (Continued)

SCHEDULED MAINTENANCE (SM)				
FUNCTION & /TASK/ SUB-TASK NUMBER	FREQUENCY	M/H'S - M/M'S	PERSONAL	ELAPSED TIME
ASSIST ENGINE SYSTEM (SM) 2.4.12		13.9 M/H'S	-	4.1 HRS.
. VISUAL	AEF	10.9 M/H'S	-	1.6 HRS.
. INSP. FOR OBVIOUS DAMAGE		1.3 M/H'S	8	10 MIN.
. INSP. FOR LEAKS		2.0 M/H'S	8	15 MIN.
. INSP. EQUIPMENT FOR SECURITY OF MOUNTING INCLUDING ENGINE MOUNTS		2.0 M/H'S	8	15 MIN.
. INSP. HOSES, LINES AND WIRE BUNDLES FOR CHAFING OR DETERIORATION		2.0 M/H'S	8	15 MIN.
. INSP. COMPRESSOR SECTION FOR DAMAGE (ROTOR AND STATOR BLADES)		20 M/M'S	1	20 MIN.
. INSP. FOR HOT SPOTS ON COMBUSTION CHAMBER SECTION		40 M/M'S	2	20 MIN.
. INSP. EXHAUST SECTION FOR CRACKS AND OBVIOUS DAMAGE		20 M/M'S	1	20 MIN.
. INSP. FOR DAMAGED TURBINE BLADES		10 M/M'S	1	10 MIN.
. CHECK OIL LEVEL		05 M/M'S	1	5 MIN.
. INSP. DEPLOYMENT MECHANISM FOR OBVIOUS DAMAGE AND SECURITY		2 M/H'S	4	30 MIN.
. FUNCTIONAL		20.0 M/H'S	-	1 HR.
. CLEAN FUEL FILTERS AND SCREENS		1 M/H	2	30 MIN.
. REPLACE IGNITER PLUGS		1 M/H	2	30 MIN.
. REPLACE ENGINE	1000 HRS.	25 M/H'S	5	5 HRS.
. UTILIZING THE ONBOARD CHECKOUT, PERFORM FULL POWER CHECK AND VERIFY PROPER READOUTS		2 M/H'S	2	1 HR.
. CLEAN OIL FILTERS		1 M/H	2	30 MIN.
. EXTEND AND RETRACT ENGINES AND CHECK TRAVEL AND PROPER MATING OF DOORS		.8 M/H	5	10 MIN.
. CHECK THROTTLE FOR PROPER RIGGING	1000 HRS.	2 M/H'S	2	1 HR.
. LEAK CHECK ENGINE WHILE PERFORMING POWER CHECK		10 MIN.	1	10 MIN.
. AFTER ENGINE OPERATION, PURGE OIL SYSTEM AND APPLY DRY LUBE TO BEARINGS.		16 M/H'S	16	1 HR.

Table 4–4
TASK ANALYSIS (Continued)

SCHEDULED MAINTENANCE (SM)

FUNCTION & /TASK/ SUB-TASK NUMBER	FREQUENCY	M/H'S - M/M'S	PERSONAL	ELAPSED TIME
CARGO TRANSFER SYSTEM (SM) 2.4.13		5.5 M/H'S	–	1.8 HRS.
. VISUAL	AEF	3.5 M/H'S	–	1.3 HRS.
. INSP. ALL VISIBLE WIRE BUNDLES FOR CHAFING, LOOSE OR BROKEN TIES AND ANCHOR CLIPS AND LOOSE OR CORRODED CONNECTORS		30 M/H'S	2	15 MIN.
. INSP. THE COMPLETE CARGO TRANSFER MECHANISM FOR OBVIOUS DAMAGE		1 M/H	2	30 MIN.
. INSP. ALL VISIBLE LINES AND HOSES FOR CHAFING, DETERIORATION AND FRAYING		30 M/H'S	2	15 MIN.
. INSP. CARGO DOORS, ACTUATOR AND MECHANISM FOR CRACKS, CORROSION, SECURITY AND DISTORTION		45 M/H'S	2	22.5 MIN.
. INSP. CARGO TRANSFER CONTROLS, ACTUATORS AND MECHANISMS FOR CRACKS, CORROSION, SECURITY AND DISTORTION		45 M/H'S	2	22.5 MIN.
. FUNCTIONAL		2 M/H'S	–	30 MIN.
. PERFORM A COMPLETE OPERATIONAL CHECK OF THE CARGO TRANSFER SYSTEM AND VERIFY PROPER OPERATION.		2 M/H'S	4	30 MIN.

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Table 4-4

TASK ANALYSIS (Continued)

SCHEDULED MAINTENANCE (SM)

FUNCTION & /TASK/ SUB-TASK NUMBER	FREQUENCY	M/H'S - M/M'S	PERSONAL	ELAPSED TIME
MATING ATTACH POINTS SYSTEM (SM) 2.4.14		2.25 M/H'S	-	45 MIN.
. VISUAL	AEF	2.25 M/H'S	-	45 MIN.
. INSP. MECHANISM FOR CLEANLINESS AND OBVIOUS DAMAGE (1ST & 2ND STAGE)		30 M/M'S	3	10 MIN.
. VISUALLY INSP. FOR LEAKAGE (HYD. OR N_2)		45 M/M'S	3	15 MIN.
. INSP. ALL VISIBLE LINES AND HOSES FOR CHAFING, DETERIORATION AND FRAYING		30 M/M'S	3	10 MIN.
. INSP. DOOR CLOSURE MECHANISMS FOR DAMAGE AND OBVIOUS BINDING OR CHAFING		30 M/M'S	3	10 MIN.

Table 4-4
TASK ANALYSIS (Continued)

SCHEDULED MAINTENANCE (SM)				
FUNCTION & /TASK/ SUB-TASK NUMBER	FREQUENCY	M/H'S - M/M'S	PERSONAL	ELAPSED TIME
HEAT PROTECTION (SM) 2.4.15		60 M/H'S	-	3 HRS.
. VISUAL	AEF	60 M/H'S	-	3 HRS.
. INSP. ALL SHINGLES AND ATTACHING PARTS FOR OBVIOUS DAMAGE (NICKS, DENTS, DISCOLORATION, HOT SPOTS)		20 N/H'S	20	1 HR.
. REMOVE AND REPLACE CRACKING SHINGLES IN THE HEAT CRITICAL AREA AND INSP. THE INSULATION AND STRUCTURE FOR DAMAGE		40 M/H'S	20	2 HRS.

Table 4-4

TASK ANALYSIS (Continued)

SCHEDULED MAINTENANCE (SM)				
FUNCTION & /TASK/ SUB-TASK NUMBER	FREQUENCY	M/H'S - M/M'S	PERSONAL	ELAPSED TIME
STRUCTURE (SM) 2.4.16 . VISUAL . PERFORM A COMPLETE VISUAL INSPECTION OF THE INNER AND OUTER ACCESSIBLE STRUCTURE. INSP. FOR CRACKS, DINGS, DENTS, GOUGES, SCRATCHES, HOT SPOTS OR ANY OBVIOUS DAMAGE OF THE FOLLOWING STRUCTURES: . NOSE CAP . FUSELAGE LEADING EDGE . EMPENNAGE FIXED SURFACE . EMPENNAGE FLAP . FINS . EMPENNAGE LEADING EDGES . WINGS . AILERONS . CABIN STRUCTURE . CABIN HATCHES . CABIN WINDOWS . MAIN INTERNAL SUPPORT STRUCTURE	AEF	180 M/H'S 180 M/H'S 160 M/H'S	- - 20	12 HRS. 12 HRS. 8 HRS.
. CLEAN COMPLETE OUTER STRUCTURE AND APPLY SURFACE COATING WHERE REQUIRED	AEF	20 M/H'S	10	8 HRS.

Table 4-4
TASK ANALYSIS (Continued)

SCHEDULED MAINTENANCE (SM)

FUNCTION & /TASK/ SUB-TASK NUMBER	FREQUENCY	M/H'S - M/M'S	PERSONAL	ELAPSED TIME
CREW STATION (SM) 2.4.17		1.6 M/H'S	–	54 MIN.
. VISUAL	AEF	1.6 M/H'S	–	54 MIN.
. INSP. ALL INDICATORS FOR DIRTY, BROKEN, SLIPPED AND SCRATCHED GLASS OR ANY OBVIOUS DAMAGE		10 M/M'S	2	5 MIN.
. INSP. ALL EQUIPMENT IN CREW STATION FOR SECURITY OF MOUNTING		10 M/M'S	2	5 MIN.
. INSP. SAFETY BELTS AND SHOULDER HARNESS FOR FRAYING OR OBVIOUS DAMAGE		02 M/M'S	2	1 MIN.
. INSP. SEAT STRUCTURE FOR CRACKS, DENTS AND OBVIOUS DAMAGE		06 M/M'S	2	3 MIN.
. INSP. INSTRUMENT PANEL FOR SECURITY OF MOUNTING		04 M/M'S	2	2 MIN.
. INSP. ALL CONTROLS FOR FREEDOM OF MOVEMENT		04 M/M'S	2	2 MIN.
. INSP. CABIN INSULATION FOR EVIDENCE OF NOT ADHERING		04 M/M'S	2	2 MIN.
. WIND THE 8-DAY CLOCK AND SET		02 M/M'S	2	1 MIN.
. INSP. ALL SWITCHES AND CIRCUIT BREAKERS FOR PROPER POSITION		10 M/M'S	2	5 MIN.
. INSP. ALL VISIBLE WIRE BUNDLES FOR CHAFING, LOOSE OR BROKEN TIES AND ANCHOR CLIPS AND LOOSE OR CORRODED TERMINALS		10 M/M'S	2	5 MIN.
. INSP. THE CREW STATION FOR CLEANLINESS		30 M/M'S	2	15 MIN.
. CLEAN THE CREW STATION INNER GLASS		20 M/M'S	2	10 MIN.

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Table 4-5

TYPES OF INSPECTION

1. Acceptance Inspection - An inspection by maintenance personnel at the maintenance area immediately following ferry flight from contractor fabrication site. This inspection consists of checking vehicle for quality of all contractor's work, inventory of publications and safety devices, and brief familiarization opportunity for personnel not previously associated with the program.

2. Postflight Inspection

   (a) Assumption - Vehicle on ramp following flight.

   (b) Definition - This inspection will be accomplished after each flight. The inspection consists of checking the vehicle to determine if it is suitable for another flight when quick turnaround is scheduled and/or determining the vehicles' status prior to going into the service area.

3. Maintenance Cycle Inspection

   (a) Assumption - Vehicle has flown and is positioned in the Maintenance Area.

   (b) Defintion - This inspection consists of checking certain components, areas, or systems of the vehicle to determine that no condition exists which would result in failure or malfunction of the component prior to the next scheduled inspection. This inspection is divided into sub-sections, numbered and will be accomplished at specified flights or engine hourly intervals. A numerical inspection is organized so each one can be accomplished in a minimum time, cover certain areas frequently and as flights or engine hours increase the corresponding higher numbered Phase Inspections will be in greater depth.

4. Special Inspections

   (a) Assumption - None.

Table 4-5

TYPE OF INSPECTION (Continued)

(b) Definition - These inspections contain requirements that will be accomplished upon the accrual of a specified number of flying hours, equipment hours of operation, a lapse of calendar time, or after the occurrence of a specific or unusual condition.

Inspections used in the development of the maintenance cylce are divided into numerical phases and defined as follows:

- Numerical phased inspections are organized such that each one can be accomplished in a minimum time, cover specified areas frequently, and increase in depth as the number of flights increases.
- Phased inspection will include a review of flight discrepancies and component malfunctions as detected and documented by the Onboard Checkout System.
- In addition inspection and checks of specified areas, components, subsystems, or systems of the vehicle will be made to determine if conditions exist that would result in a failure or malfunction prior to the next scheduled inspection.
- Equipment that has malfunctioned or that will exceed service life will be removed and replaced.
- Equipment will be replaced with new or recertified equipment from material control.
- After equipment replacement, the system will undergo a functional check using the Onboard Checkout System to verify and document system integrity.
- Following the functional checks of the repaired systems, the vehicle

TYPE OF INSPECTION (Continued)

will undergo an integral systems test using the Onboard Checkout System to verify that the vehicle systems perform within specification limits and that systems integration is complete.

- At the completion of integral systems test, the vehicle will be "closed out" by retracting the wings and assist engines, closing and installing all hatches and doors, and installing the heat protection panels that were removed for access.
- Following completion of vehicle "close out", the vehicle will be prepared for moving.
- Phased inspection operation will restore the vehicle to mission ready condition with minimum effort.

MAINTENANCE TIMELINE ANALYSIS

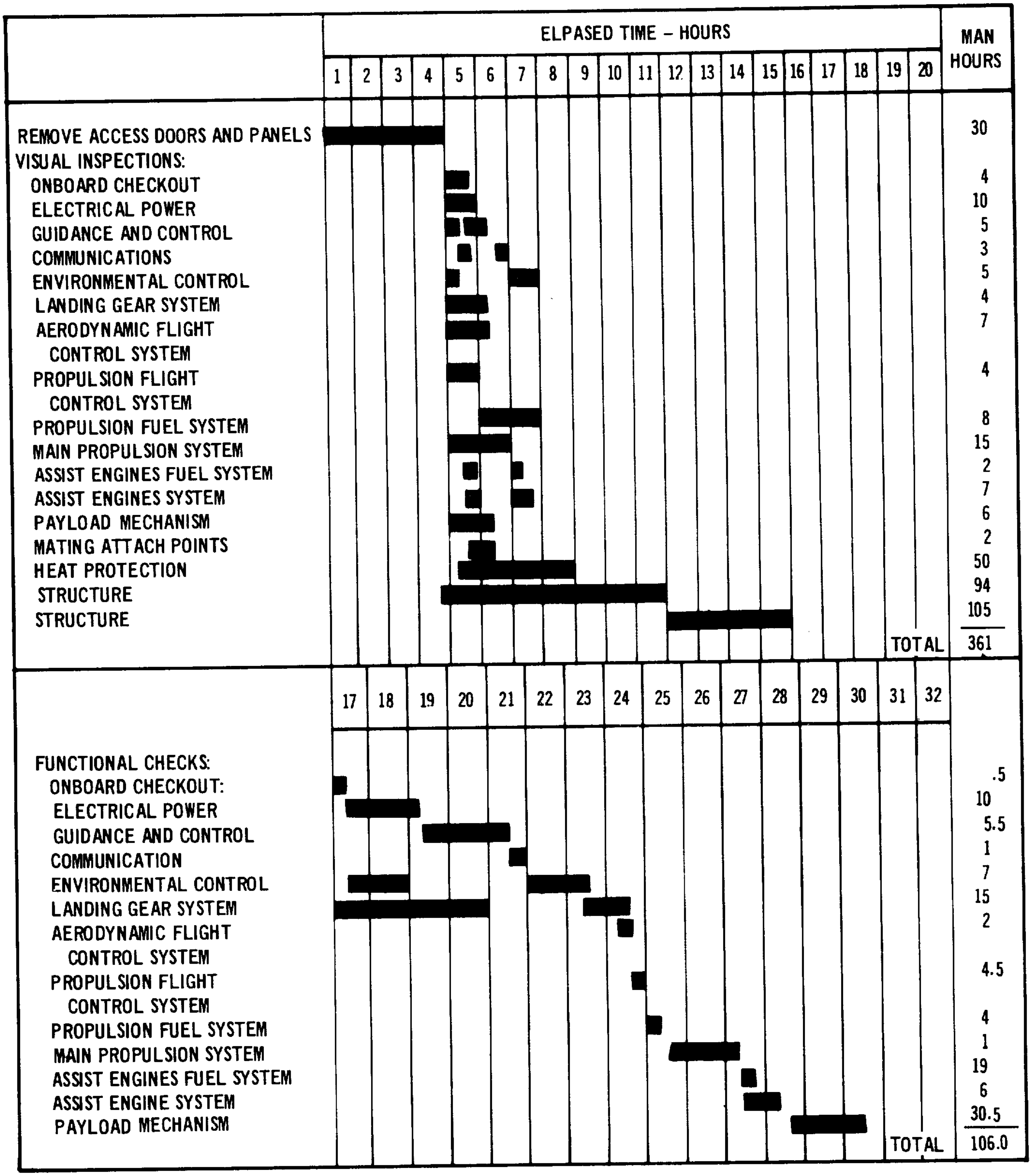


Figure 4-7

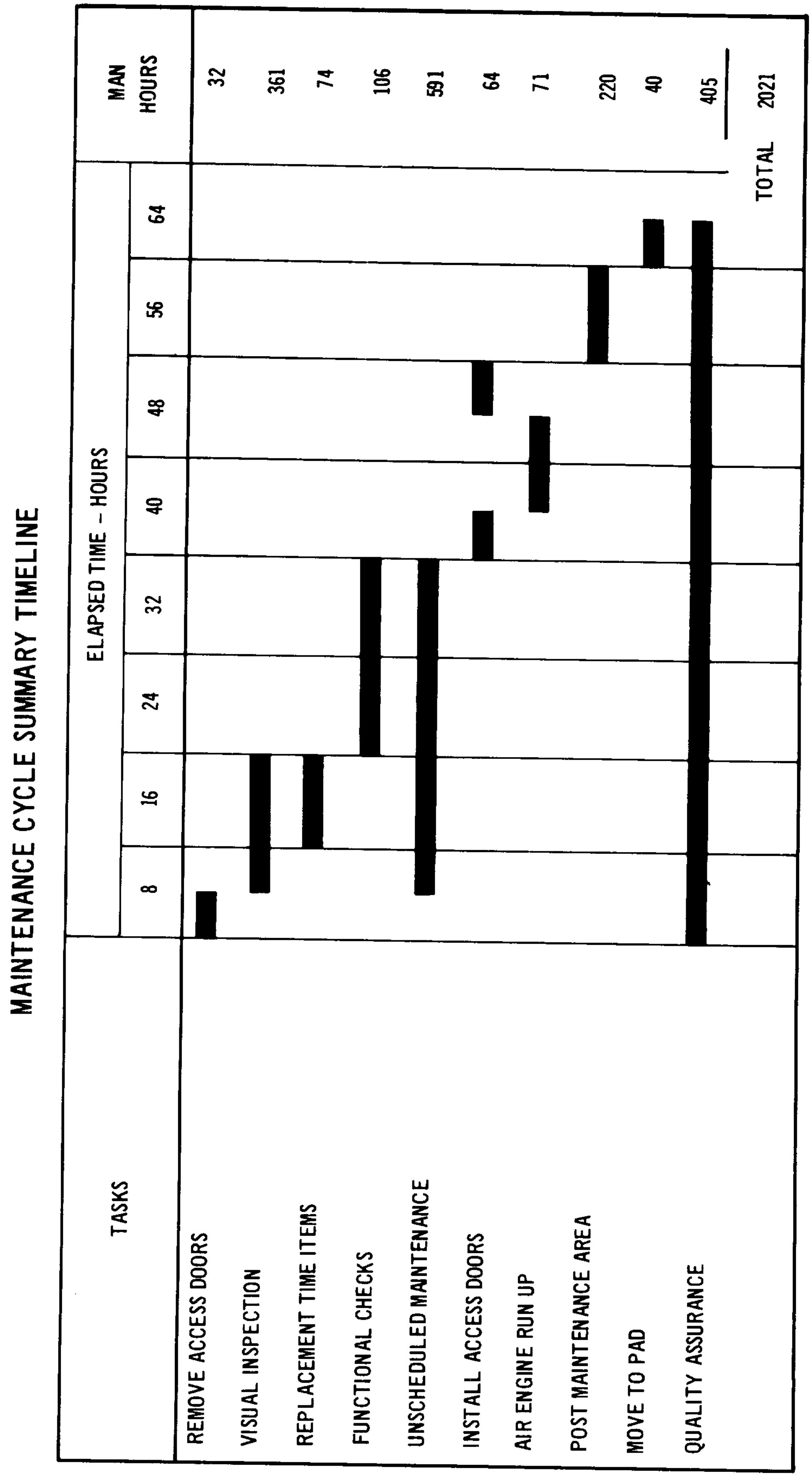

MAINTENANCE CYCLE SUMMARY TIMELINE
TASKS
ELAPSED TIME - HOURS
8
16
24
32
40
48
56
64
MAN HOURS
REMOVE ACCESS DOORS
32
VISUAL INSPECTION
361
REPLACEMENT TIME ITEMS
74
FUNCTIONAL CHECKS
106
UNSCHEDULED MAINTENANCE
591
INSTALL ACCESS DOORS
64
AIR ENGINE RUN UP
71
POST MAINTENANCE AREA
220
MOVE TO PAD
40
QUALITY ASSURANCE
405
TOTAL
2021

Figure 4-8

Figure 4-7. A maintenance summary timeline for either the Carrier or Orbiter is presented in Figure 4-8.

Manpower Analysis - Figures 4-9, 4-10, and 4-11 illustrate the manpower spread needed to perform the detailed functions of post flight maintenance, and visual and functional checks at the maintenance area during the maintenance cycle. The analysis included staffing for each function.

A further breakdown of vehicle turnaround manpower utilization was derived from this analysis. The additional manpower functions required for the turn-around activity are:

- Corrective Maintenance
- Servicing
- Payload Installation
- Air Breathing Engine Run Up
- Door Removal and Installation

The analyses indicated that complete vehicle turnaround activity (carrier and orbiter) required approximately 180 personnel per shift for two (2) shifts to perform maintenance tasks and approximately 120 personnel per shift for three (3) shifts (24 hours) to perform launch preparation. The same skill level is required for maintenance and launch operations.

In conclusion, approximately 360 personnel are required to support the turnaround activities of the carrier and orbiter. This complement of 360 includes direct, indirect, and administration personnel. The manhours required for the turnaround are identified in Table 4-6. The administration time factor consist of time in nontechnical routines (e.g., sickness, personal time, etc.).

4.1.3 Launch Preparation

Summary - Upon completion of the maintenance cycle the vehicles are ready to enter the launch preparation phase, which is the final 24-hour period prior to launch. Results of the launch preparation analysis are indicated in Table 4-7.

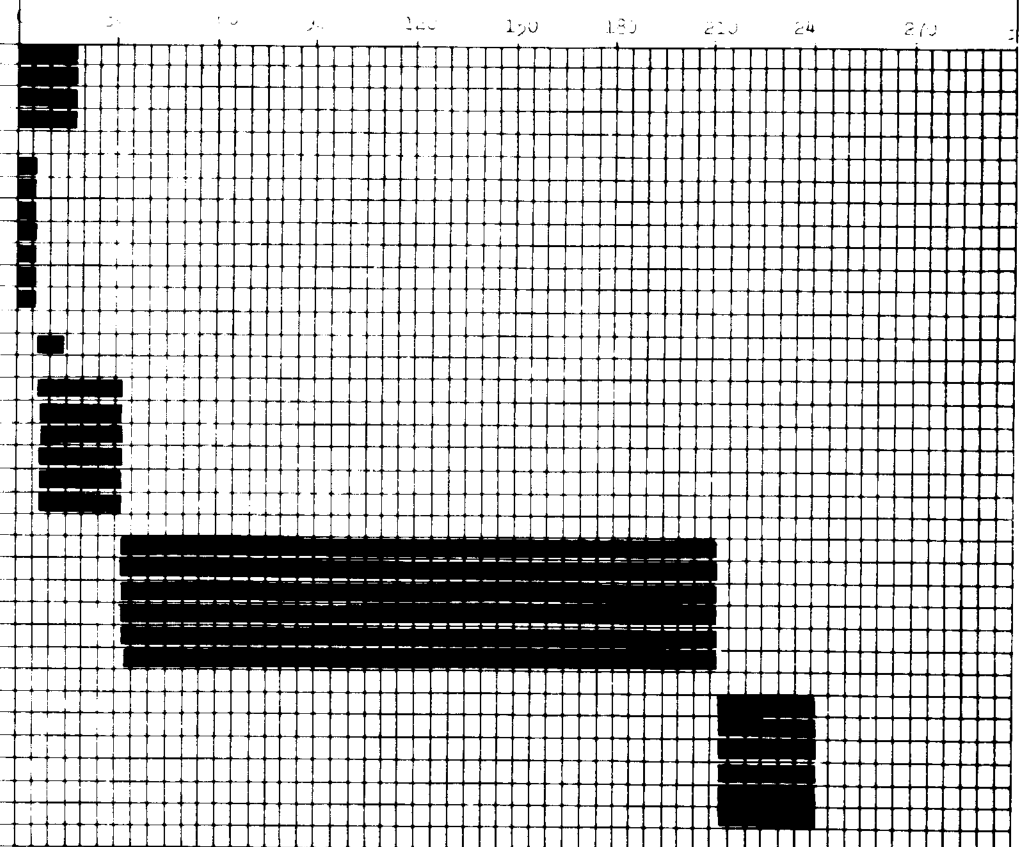
The purpose of this analysis is to identify the requirements and constraints for the launch preparation cycle and to establish facilities and equipment

POST FLIGHT MANPOWER STAFFING

FUNCTION	LOCATION	TYPE OF MAINTENANCE (IF APPLICABLE)
POST FLIGHT MAINTENANCE	MAINTENANCE AREA	POST FLIGHT

SOURCE OF FUNCTION	FUNCTION AND CORRESPONDING TASKS (IF APPLICABLE)
	CREW EGRESS & DATA REMOVAL
	INSTALL SAFETY DEVICES
	CABIN SWITCH CHECK
	WALK AROUND VISUAL, QA INSP.
	COOL VEHICLE
	MOVE VEHICLE TO SERVICE AREA

TIME (MINUTES)



Note: Each = (1) one man

TIME LINE SHEET

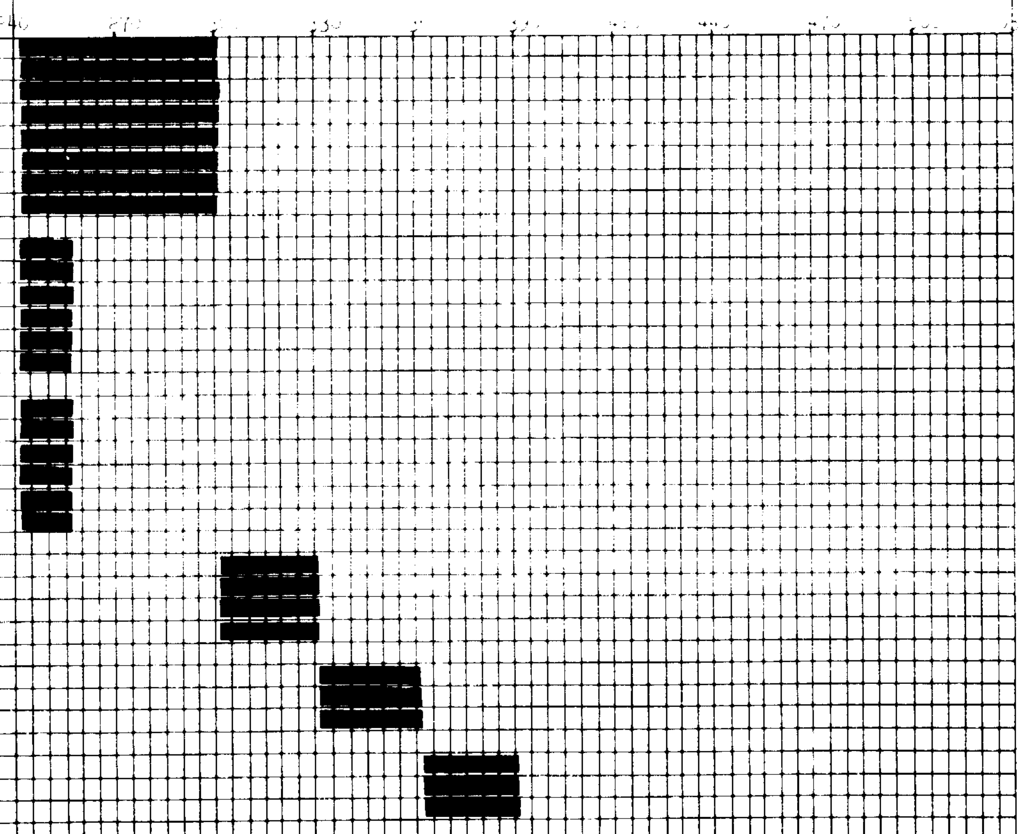
Sheet 1 of 4

Figure 4-9

POST MANPOWER STAFFING (Continued)

FUNCTION	LOCATION	TYPE OF MAINTENANCE (IF APPLICABLE)
POST FLIGHT MAINTENANCE	MAINTENANCE AREA	POST FLIGHT (Continued)

SOURCE OF FUNCTION	FUNCTION AND CORRESPONDING TASKS (IF APPLICABLE)	TIME (MINUTES)
	REMOVE PAYLOAD	
	POSITION EMERGENCY EQUIPMENT	
	INVENTORY & REMOVE REQUIRED EQUIPMENT	
	PROVIDE ACCESS TO REQUIRED AREAS	
	HOOK UP DESERVICING OMACS AGE	
	DESERVICE OMACS PRESSURANT	



Note Each ▬ = (1) one man

TIME LINE SHEET

Sheet 2 of 4

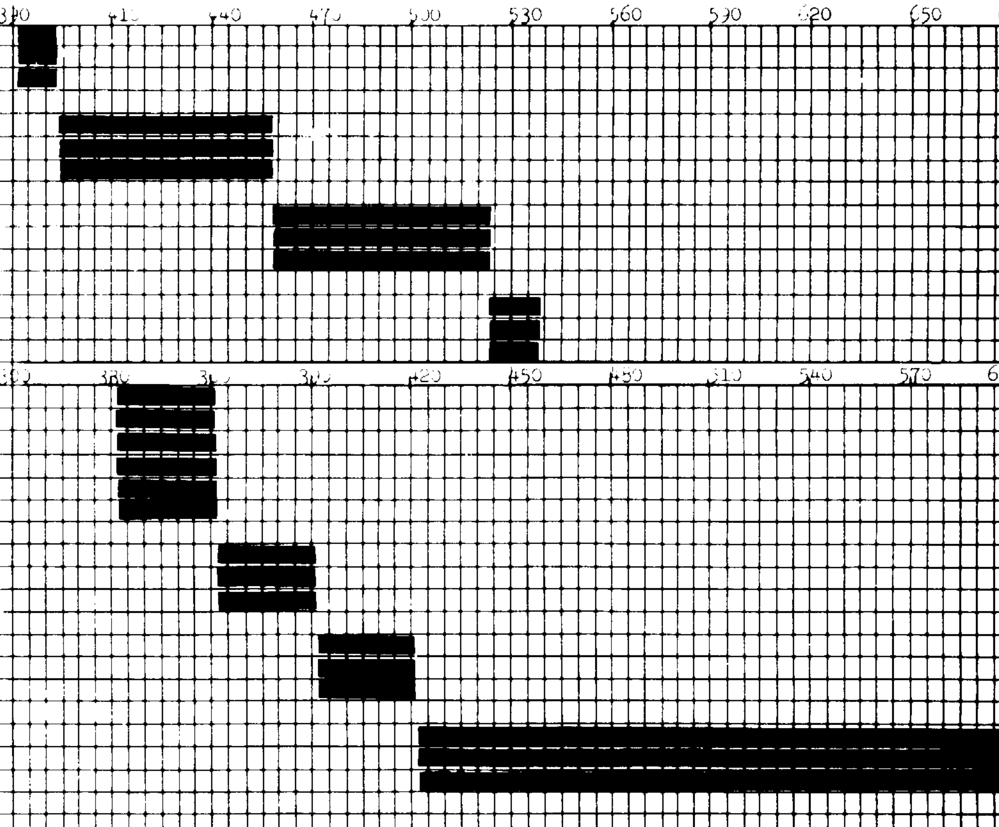
Figure 4-9

POST MANPOWER STAFFING (Continued)

FUNCTION	LOCATION	TYPE OF MAINTENANCE (IF APPLICABLE)
POST FLIGHT MAINTENANCE	MAINTENANCE AREA	POST FLIGHT (Continued)

SOURCE OF FUNCTION	FUNCTION AND CORRESPONDING TASKS (IF APPLICABLE)
	DESERVICE HIGH PRESSURE ACCUMULATORS
	DESERVICE OMACS OXIDIZER
	DESERVICE OMACS FUEL
	PERFORM POST DESERVICING PROCEDURE
	HOOK UP DESERVICING MAIN PROPULSION AGE
	DESERVICE PROPULSION PRESSURANT
	DESERVICE PNEUMATIC CONTROL SUBSYSTEM
	DESERVICE OXIDIZER SUBSYSTEM

TIME (MINUTES)



(1) one man

TIME LINE SHEET

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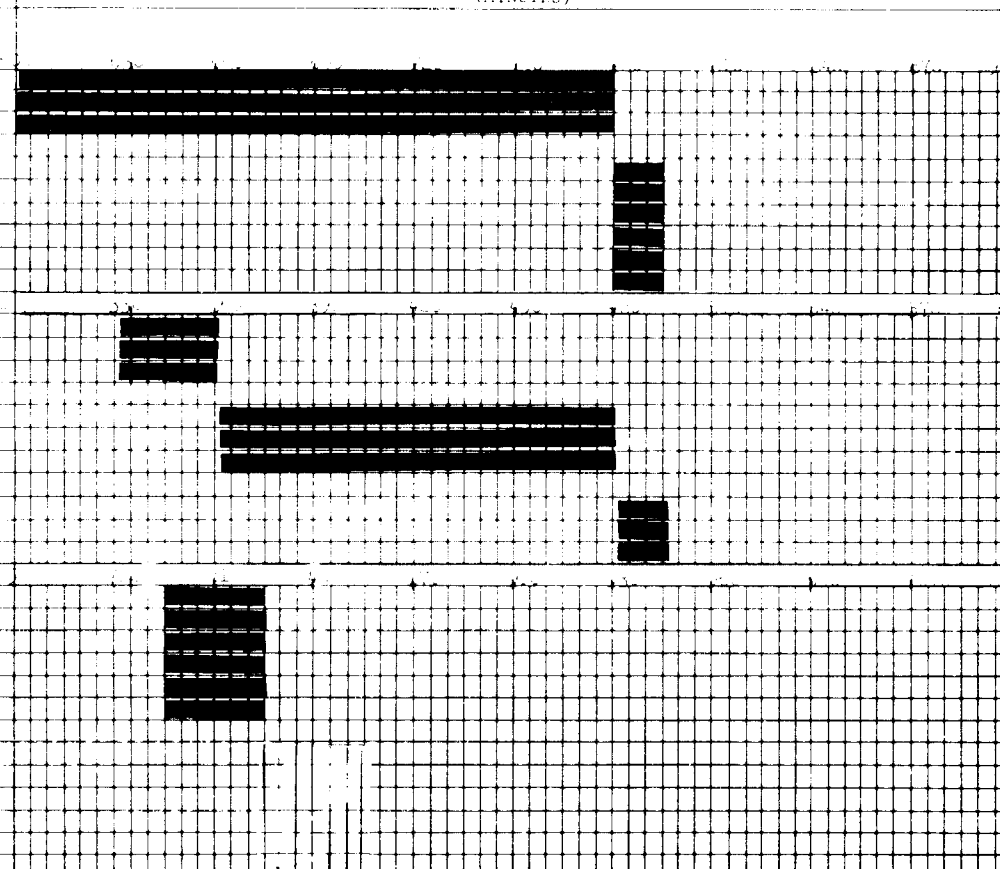
Figure 4–9

POST MANPOWER STAFFING (Continued)

FUNCTION	LOCATION	TYPE OF MAINTENANCE (IF APPLICABLE)
POST FLIGHT MAINTENANCE	MAINTENANCE AREA	POST FLIGHT (Continued)

SOURCE OF FUNCTION	FUNCTION AND CORRESPONDING TASKS (IF APPLICABLE)
	DESERVICE FUEL SUBSYSTEM
	PERFORM POST DESERVICING
	HOOK UP FUEL CELL DESERVICING AGE
	DESERVICE FUEL CELL SUBSYSTEM
	PERFORM POST DESERVICING PROCEDURE
	MOVE TO MAINTENANCE AREA

TIME (MINUTES)



Note Each ▬ (1) one man

TIME LINE SHEET

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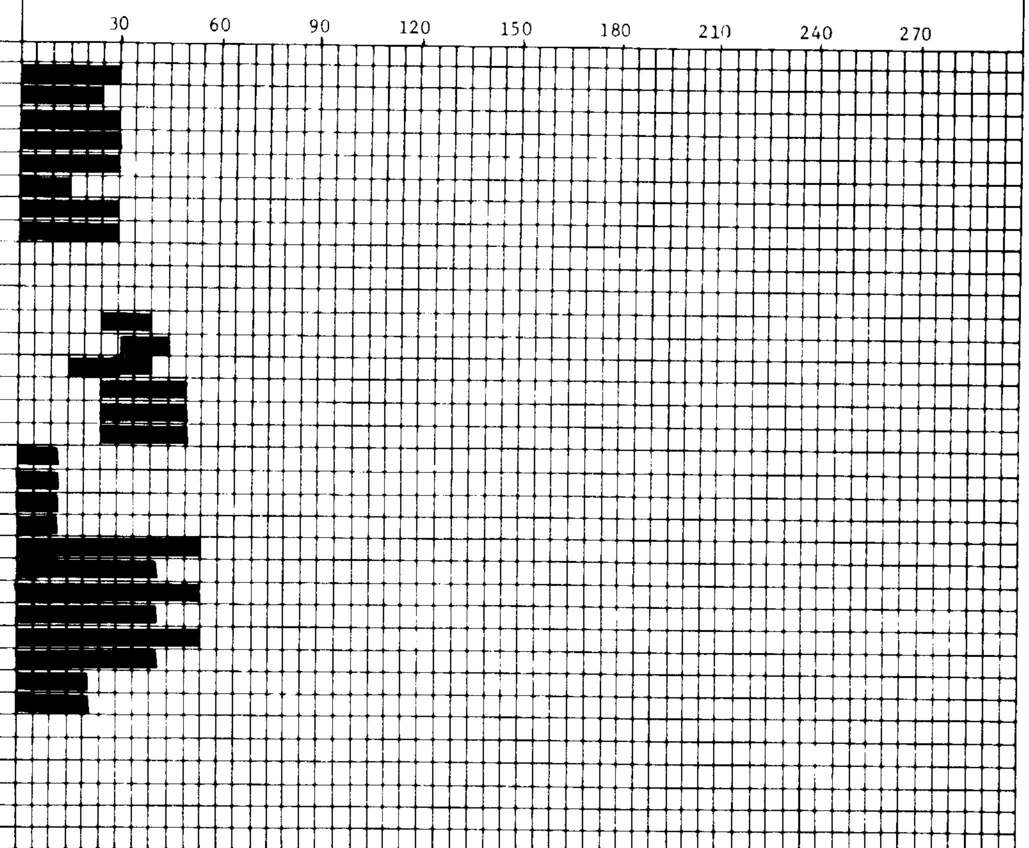
Figure 4-9

SCHEDULED MAINTENANCE MANPOWER STAFFING (VISUAL)

FUNCTION	LOCATION	TYPE OF MAINTENANCE (IF APPLICABLE)
SCHEDULED MAINTENANCE	MAINTENANCE AREA	VISUAL

SOURCE OF FUNCTION	FUNCTION AND CORRESPONDING TASKS (IF APPLICABLE)	TIME (MINUTES)
	ZONE	
	ONBOARD CHECKOUT SYSTEM	
	ELECTRICAL POWER SYSTEM	



NOTE: EACH ▬ = (1) ONE MAN

TIME LINE SHEET

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Figure 4-10

SCHEDULED MAINTENANCE MANPOWER STAFFING (VISUAL) (Continued)

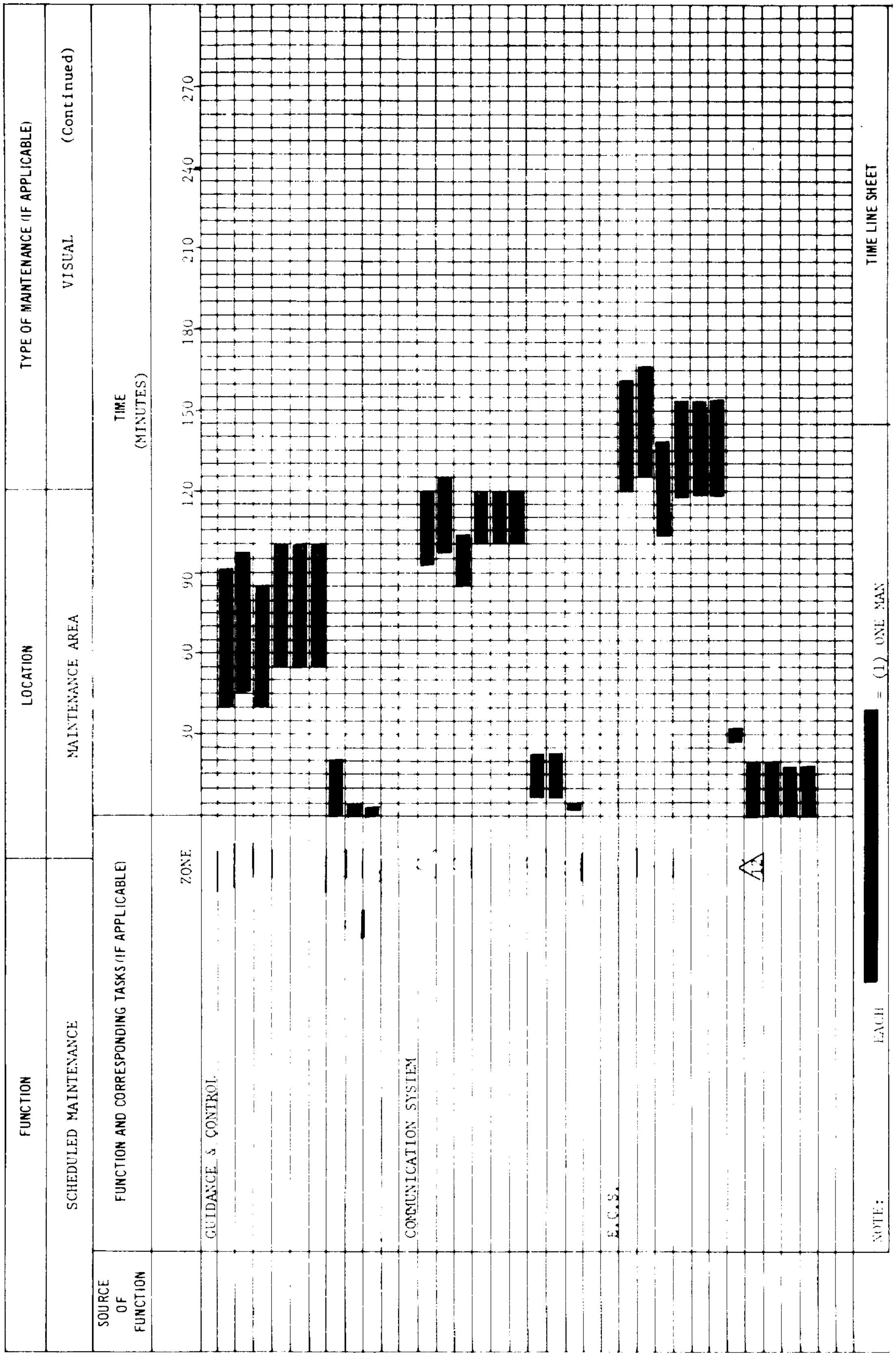

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Figure 4-10

SCHEDULED MAINTENANCE MANPOWER STAFFING (VISUAL) (Continued)

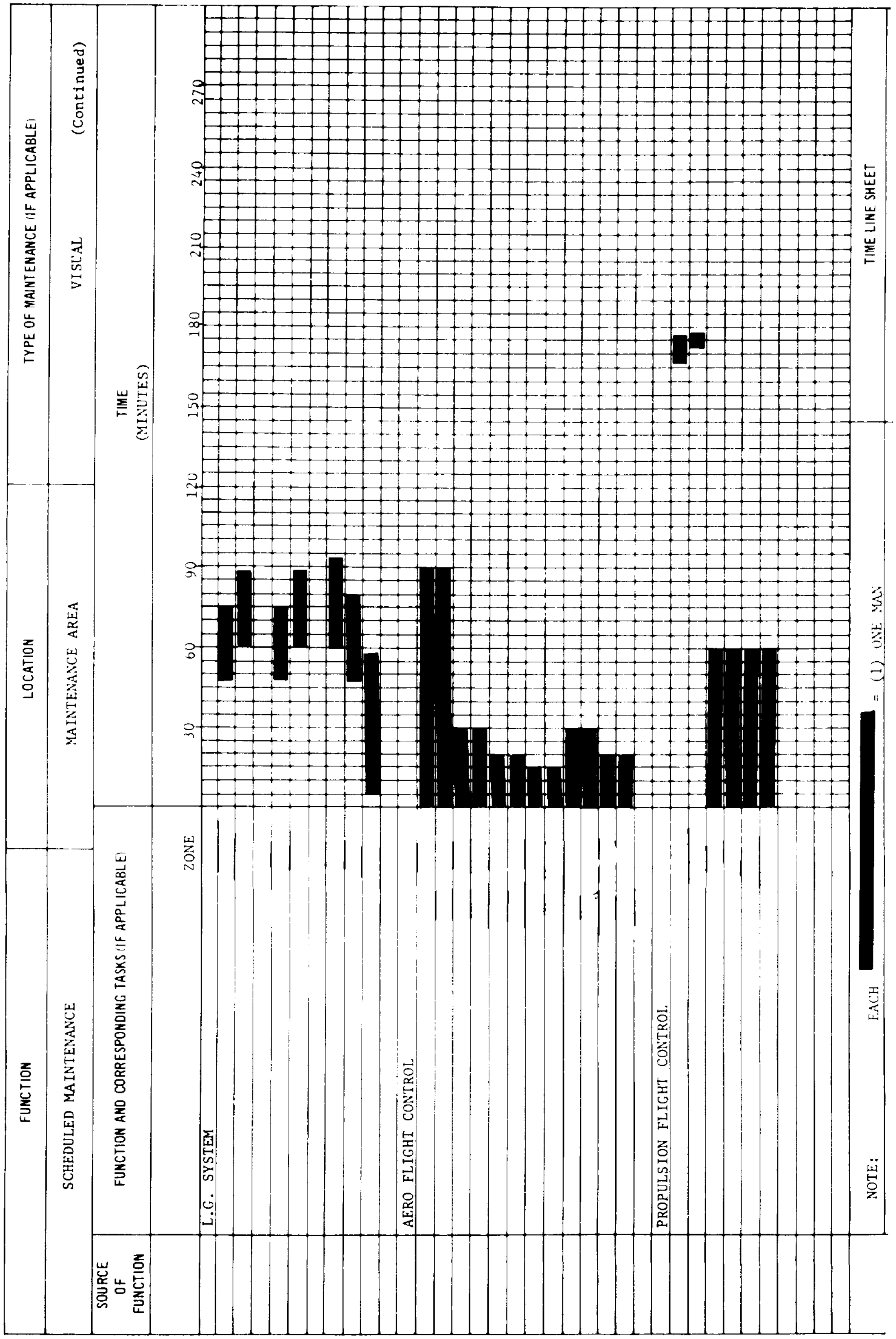

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Figure 4-10

SCHEDULED MAINTENANCE MANPOWER STAFFING (VISUAL) (Continued)

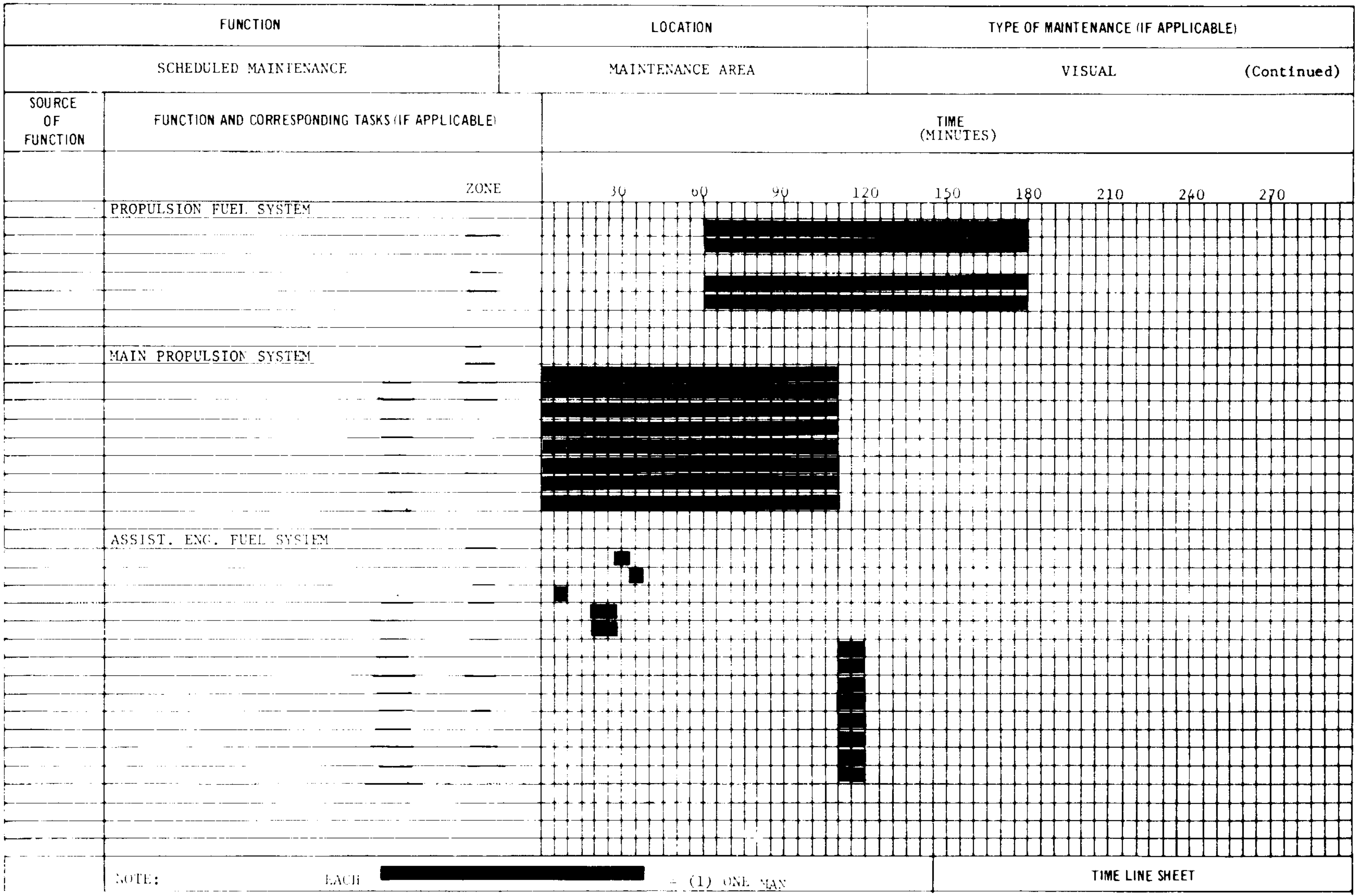

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Figure 4-10

SCHEDULED MAINTENANCE MANPOWER STAFFING (VISUAL) (Continued)

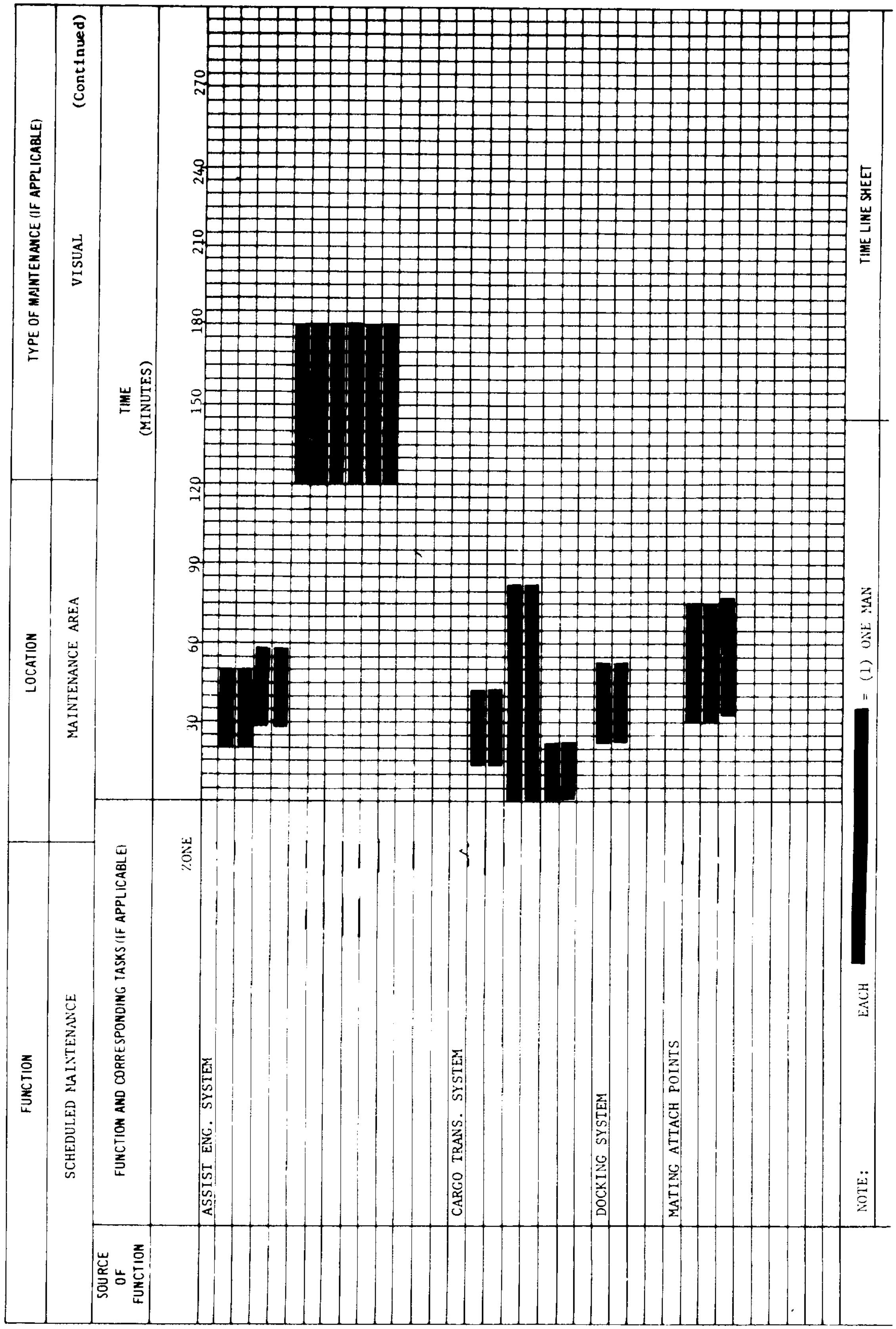

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Figure 4-10

SCHEDULED MAINTENANCE MANPOWER STAFFING (VISUAL) (Continued)

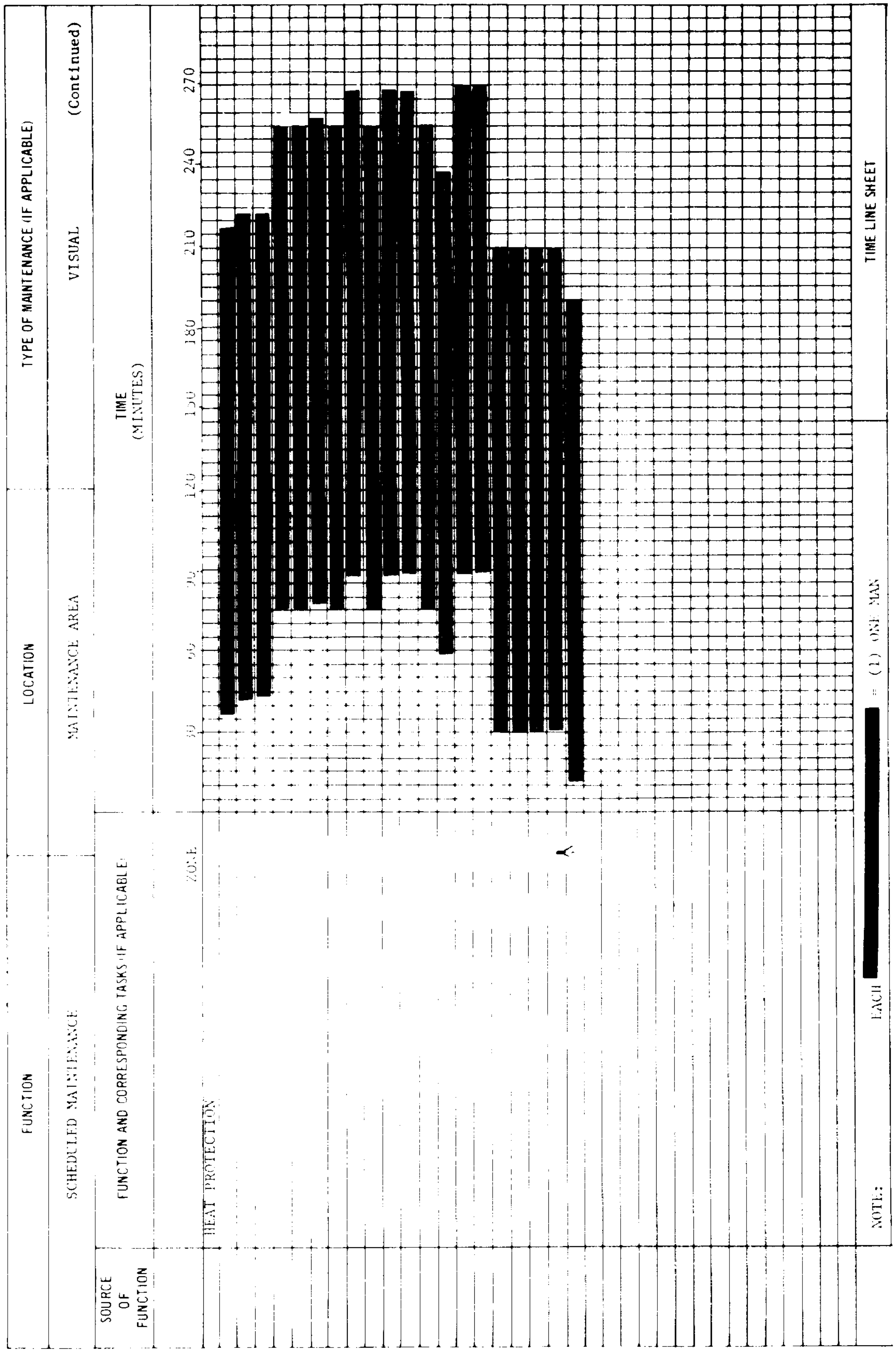

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Figure 4-10

SCHEDULED MAINTENANCE MANPOWER STAFFING (VISUAL) (Continued)

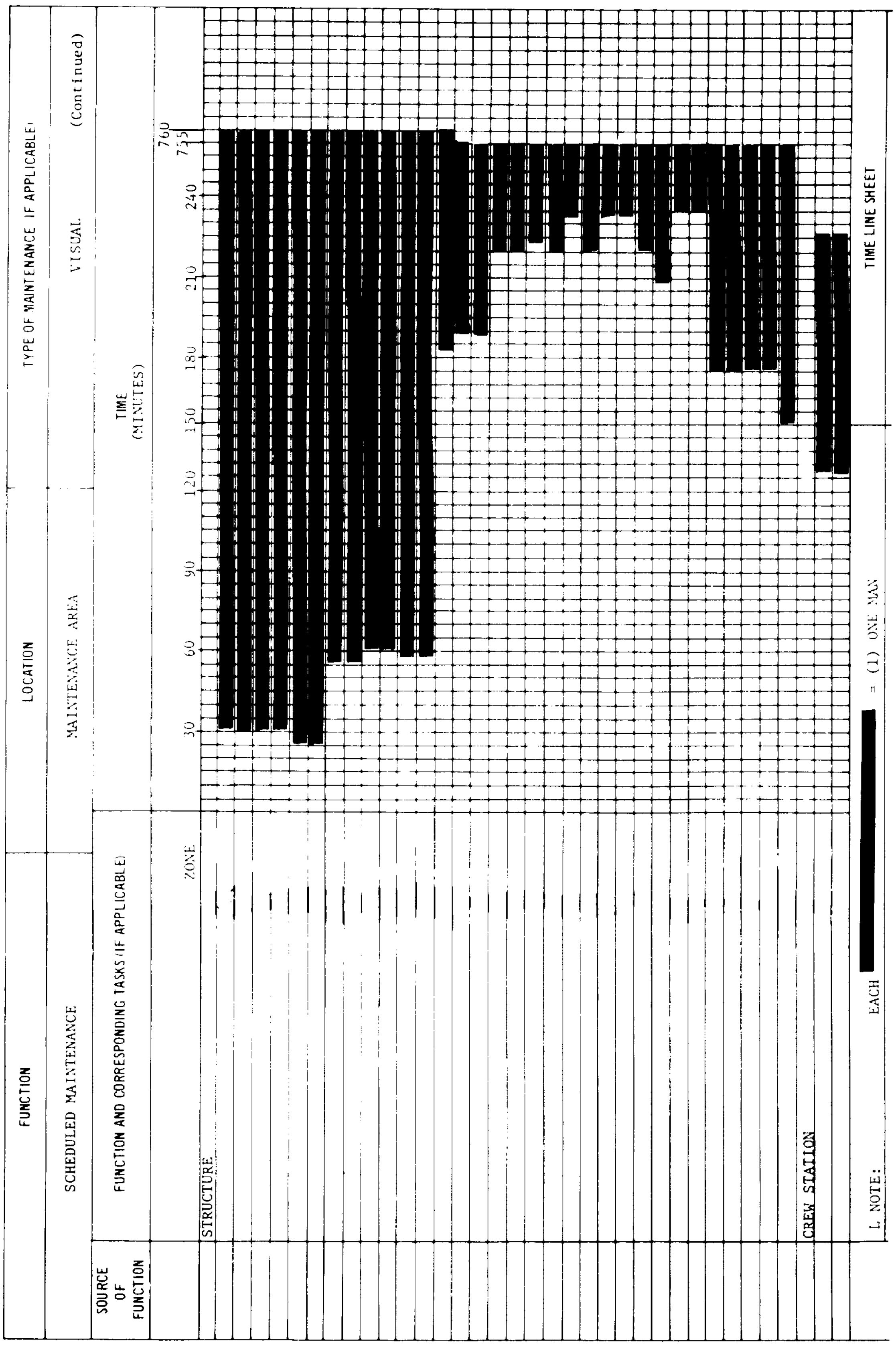

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Figure 4-10

SCHEDULED MAINTENANCE MANPOWER STAFFING (FUNCTIONAL)

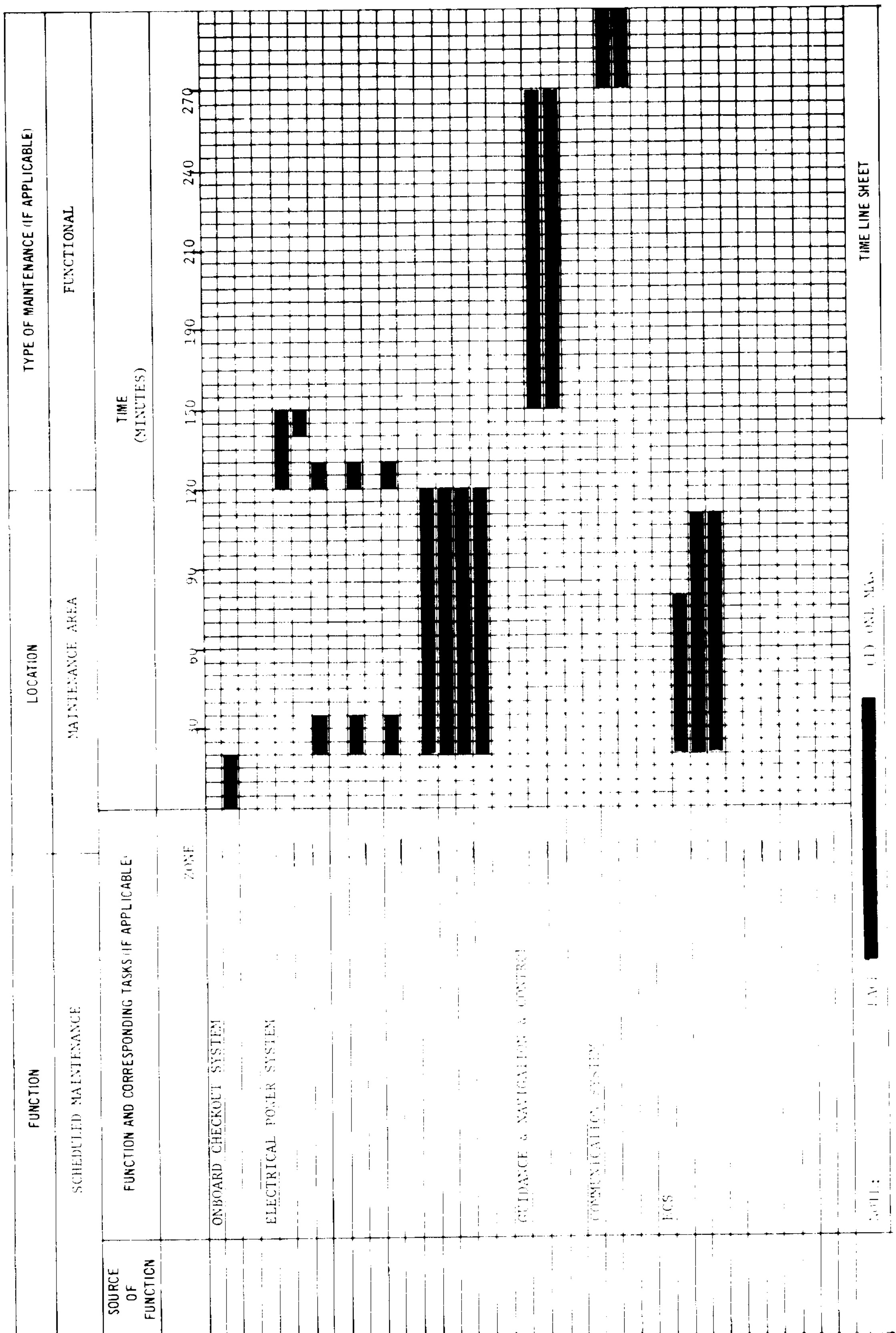

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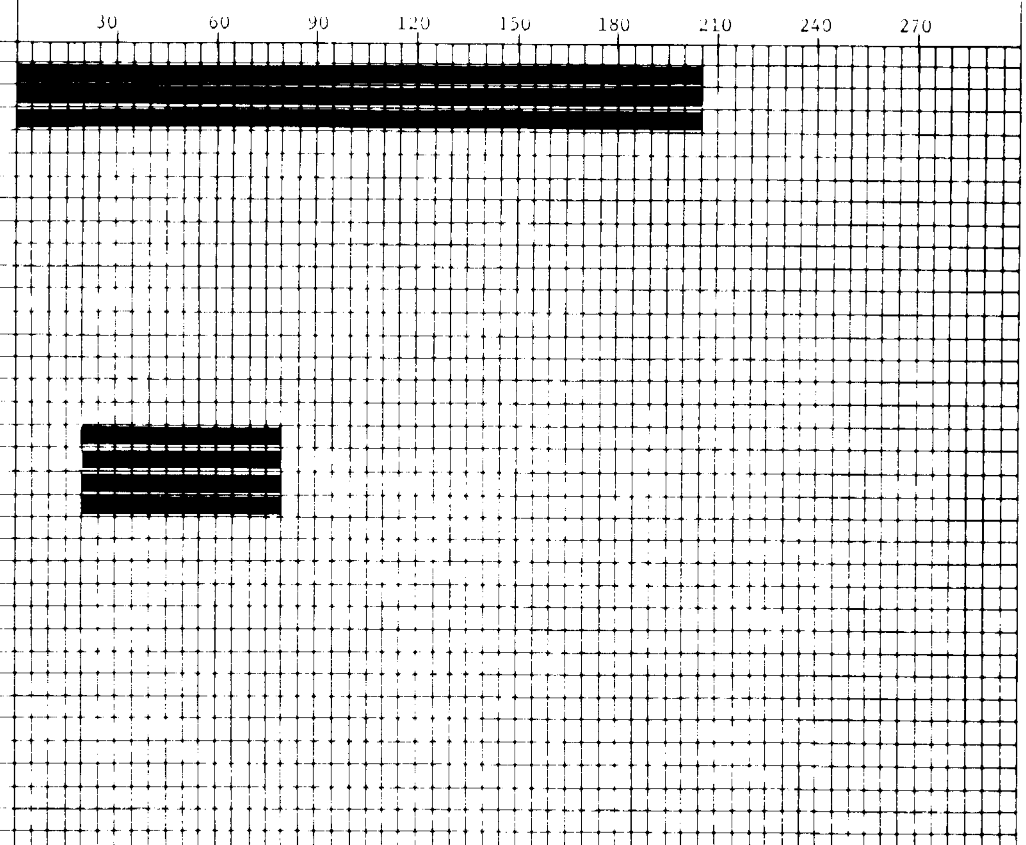
Figure 4-11

SCHEDULED MAINTENANCE MANPOWER STAFFING (FUNCTIONAL) (Continued)

FUNCTION	LOCATION	TYPE OF MAINTENANCE (IF APPLICABLE)
SCHEDULED MAINTENANCI	MAINTENANCE AREA	FUNCTIONAL (Continued)

SOURCE OF FUNCTION	FUNCTION AND CORRESPONDING TASKS (IF APPLICABLE)	TIME (MINUTES)
	LANDING GEAR SYSTEM	
	PROPULSION FLIGHT CONTROL SYSTEM	



NOTE: EACH ▬ = (1) ONE MAN

TIME LINE SHEET

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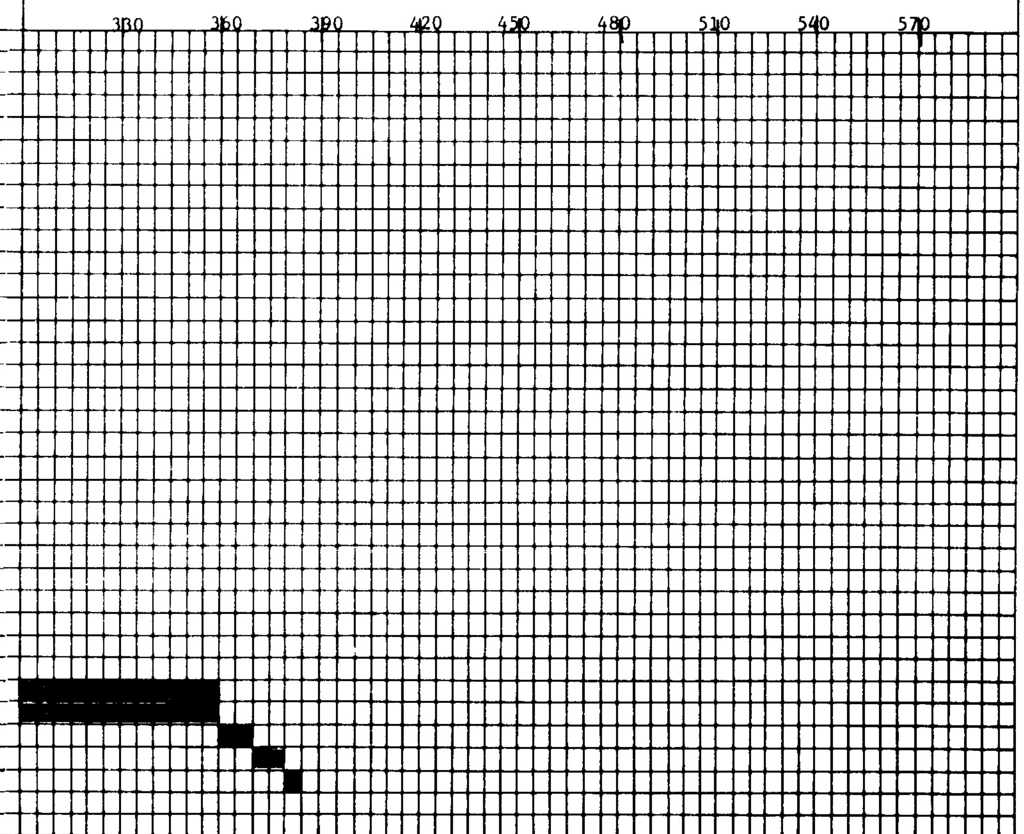
Figure 4-11

SCHEDULED MAINTENANCE MANPOWER STAFFING (FUNCTIONAL) (Continued)

FUNCTION	LOCATION	TYPE OF MAINTENANCE (IF APPLICABLE)
SCHEDULE MAINTENANCE	MAINTENANCE AREA	FUNCTIONAL (Continued)

SOURCE OF FUNCTION	FUNCTION AND CORRESPONDING TASKS (IF APPLICABLE)
	ONBOARD CHECKOUT SYSTEM
	ELECTRICAL POWER SYSTEM
	GUIDANCE & NAVIGATION & CONTROL
	COMMUNICATION SYSTEM
	ECS

TIME (MINUTES)



NOTE: EACH ▬ = (1) ONE MAN

TIME LINE SHEET

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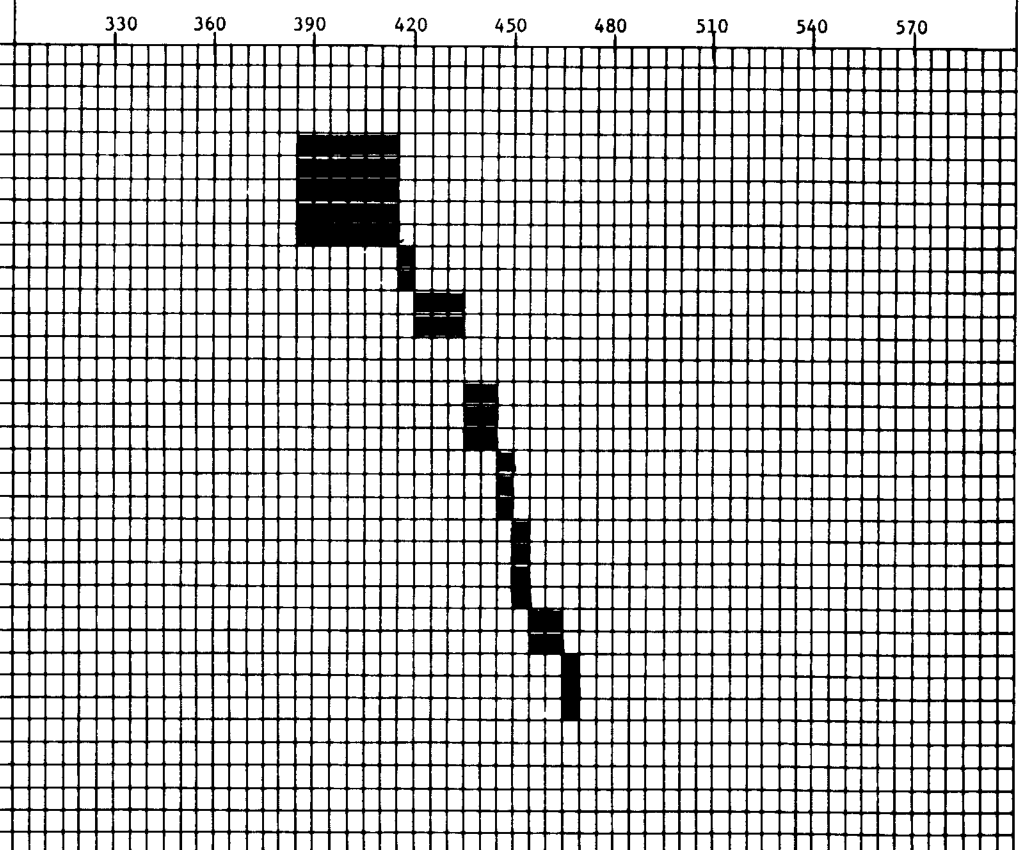
Figure 4-11

SCHEDULED MAINTENANCE MANPOWER STAFFING (FUNCTIONAL) (Continued)

FUNCTION	LOCATION	TYPE OF MAINTENANCE (IF APPLICABLE)
SCHEDULED MAINTENANCE	MAINTENANCE AREA	FUNCTIONAL (Continued)

SOURCE OF FUNCTION	FUNCTION AND CORRESPONDING TASKS (IF APPLICABLE)	TIME (MINUTES)
	LANDING GEAR SYSTEM	
	AERO FLIGHT CONTROL SYSTEM	



NOTE: EACH = (1) ONE MAN

TIME LINE SHEET

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Figure 4-11

SCHEDULED MAINTENANCE MANPOWER STAFFING (FUNCTIONAL) (Continued)

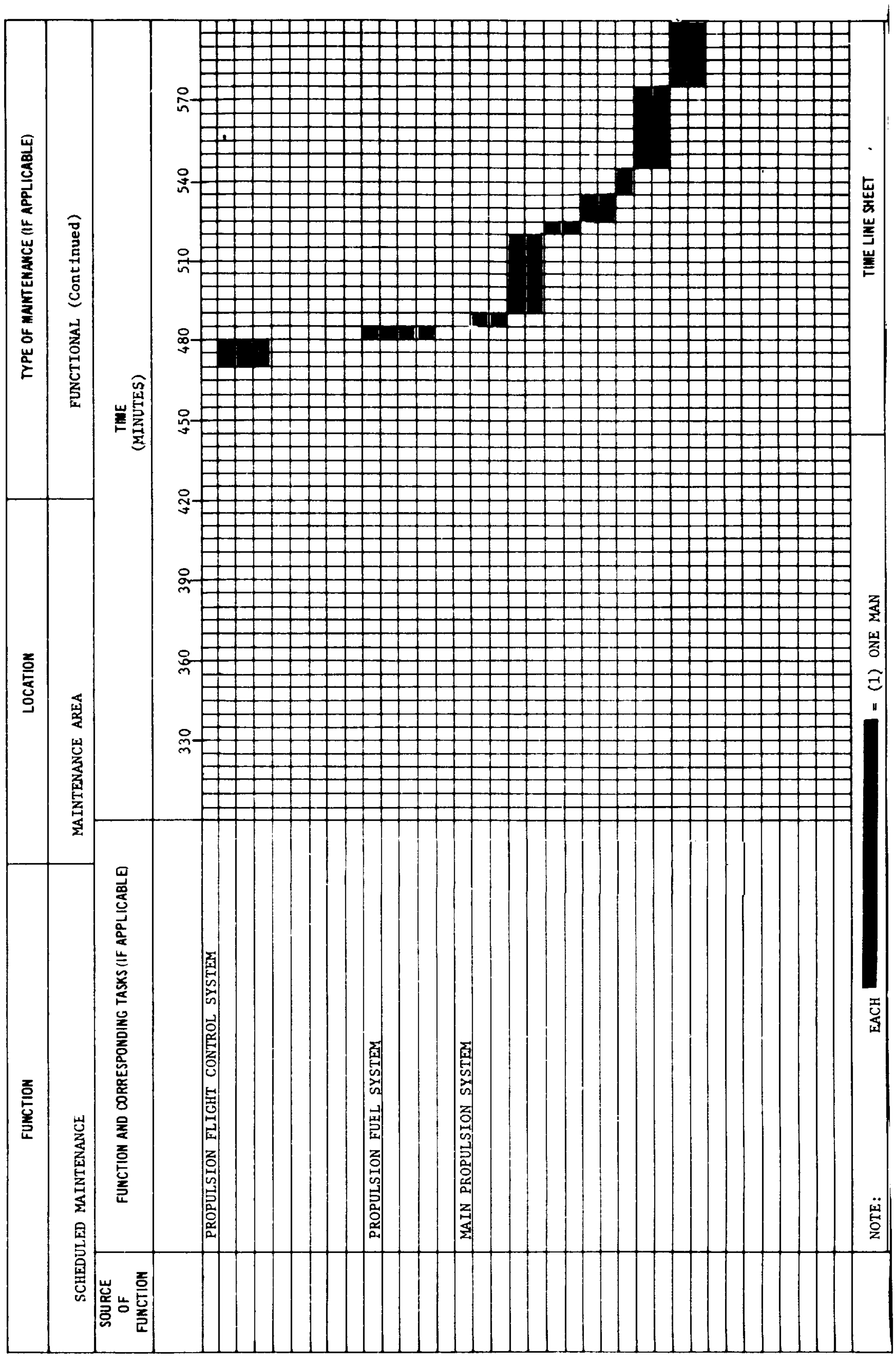

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Figure 4-11

SCHEDULED MAINTENANCE MANPOWER STAFFING (FUNCTIONAL) (Continued)

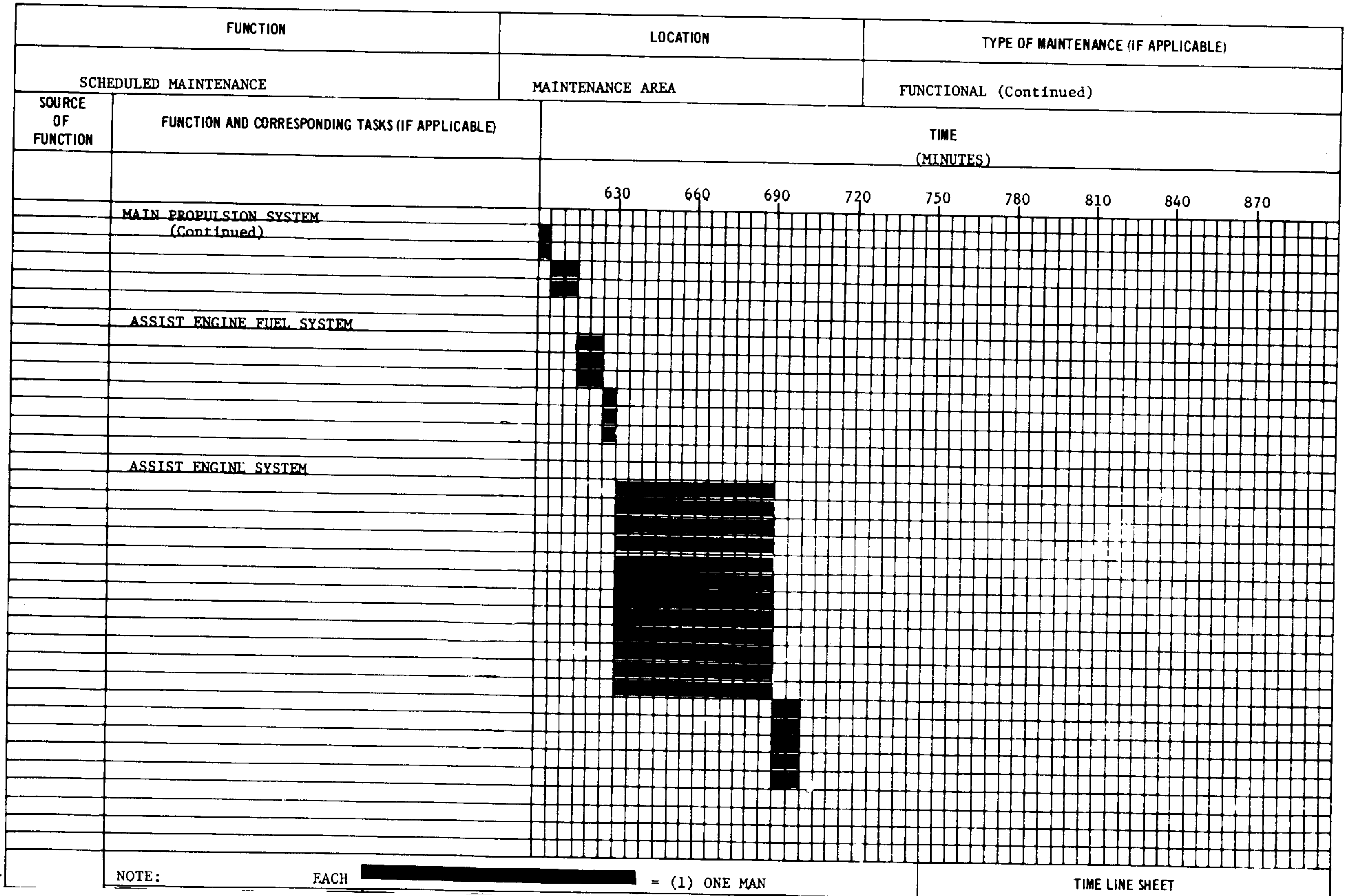

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Figure 4-11

SCHEDULED MAINTENANCE MANPOWER STAFFING (FUNCTIONAL) (Continued)

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Figure 4-11

Table 4-6

VEHICLE TURNAROUND TIME MANHOURS

FUNCTION	MANHOURS
Post Flight	186
Maintenance Cycle	4042
Launch Preparation	856
Total Direct Manhours	5084
Indirect Manhours at 200%	10,164
Total Direct/Indirect Manhours	15,248
Administrative at 12%	1,830
Total Turnaround Manhours	17,078

Table 4-7

WHAT WAS LEARNED FROM LAUNCH PREPARATION ANALYSIS

- o Launch pad schedule limited to tasks that cannot be performed in advance.
  - o Reason - Retaining the vehicles in a horizontal position until just prior to launch enhances the access to the vehicles.
- o Advantages to utilizing the VAB for Pre-Pad erection.
  - o Reason - Maximum use of existing facilities; VAB could be used for the maintenance cycle; no erectors required in the launch pad area; can checkout integrated system before going to the pad.
- o Disadvantage of using VAB
  - o Reason - A field splice of the first stage vehicle wing tips required to enter the high bay cell.
- o Only 24 hours on-the-pad is required
  - o Reason - Servicing and final system checkout necessary before launch will closely parallel activities required to prepare commercial airlines for flight making maximum use of onboard checkout with minimum ground support.

requirements. Basically, two techniques using existing equipment are considered feasible for vehicle erection and integration. These techniques are the prepad using the Vertical Assembly Building (VAB) and Launch Umbilical Tower (LUT) and on-pad, using an existing launch pad requiring a new erector and tower structure. The same basic activities are accomplished in either technique, the only difference being in where the activities are performed.

Erection - The Pre-Pad technique, shown in Figures 2-28 through 2-32 in Section 2.4 of Volume III, would require a high bay area and crane capability to translate each vehicle from the horizontal to vertical position. The Carrier would be erected first using a crane and dolly. The vehicle is raised in the vertical position, a launcher placed beneath it, and the vehicle secured to the launcher. The second stage, with payload installed, is erected in the same manner after which the vehicles are mated. After checking the vehicle system for system compatibility the mobile launcher is moved to the launch pad. Moving the launcher to the pad and connecting the LUT to the pad facilities is estimated to take 12 hours.

Using the On-Pad technique, shown in Figures 2-33 and 2-34 in Section 2.4 of Volume III, individually in the horizontal position to the launch pad. The Carrier would be raised with an erection device which would be built in a pit in the concrete ramp. The second stage would then be erected using another erection device and the vehicles would be mated. Using this technique would require about 13 hours including moving both stages to the pad, erecting each stage and integrating the system. (NOTE: Complete explanations of the Pre-Pad and On-Pad techniques are included in the launch operations plan, Volume III, Section 2.4.

On-Pad Operations - Following either the Pre-Pad or the On-Pad erection and integration, the remaining On-Pad operations are applicable to either technique and include:

- o Hook-up and checkout of fluid and gas connections.
- o Power-up and check range and navigation inputs.
- o JP-4 fueling and propulsion system operational checkout.
- o Final launch preparation and inspection.
- o Crew exit and cryogenic servicing.
- o Crew and passenger boarding.
- o Final systems checkout utilizing OCS.
- o Terminal countdown
- o Launch

4.2 Mission Interfaces and Payload Handling/Accommodations - The major subtasks of the Mission Interfaces and Payload Handling/Accommodations Special Emphasis Study are outlined in the task flow diagram shown in Figure 4-12. Specifically, these subtasks are:

- o Mission interface definition
- o Mission interface impact assessment
- o Payload-handling facility identification
- o Passenger-handling facility identification
- o Parametric analyses with mission duration
- o Design implications drawn from parametrics

The following paragraphs address themselves to each of these subtasks and present the study approach and the more significant results which emerged from the investigation.

4.2.1 Mission Interface Definition - In order to insure the identification of all major mission interfaces, mission interface definition was performed on a mission-event basis. With this approach, each mission event and/or proposed mission capability was explored to determine available alternate approaches, and from these approaches the preferred alternatives were selected. A summary of the mission interface definitions is given in Table 4-8. The preferred mode of operation for each mission event is indicated by a box.

4.2.2 Mission-Interface Impact Assessment - The greatest asset of the foregoing approach to the definition of the mission interfaces is that it immediately forces the investigator to choose a rationalized, coherent chain of preferred events from mission start to mission end. It also provides the reviewer with in-depth visibility of the interplay among the various mission events and the mission systems. By not discarding alternate approaches when a preferred approach is selected, further changes in the preferred mission are more easily made downstream in the study. One final advantage of this method is that it points out areas where further investigation is warranted and/or trade studies should be performed. By identifying these areas early in the study, much wasting and/or duplication of effort can be avoided.

In identifying the mission interfaces of Table 4-8, five major problem areas concerning payload transfer surface immediately:

MISSION INTERFACES AND CARGO ACCOMMODATIONS/HANDLING

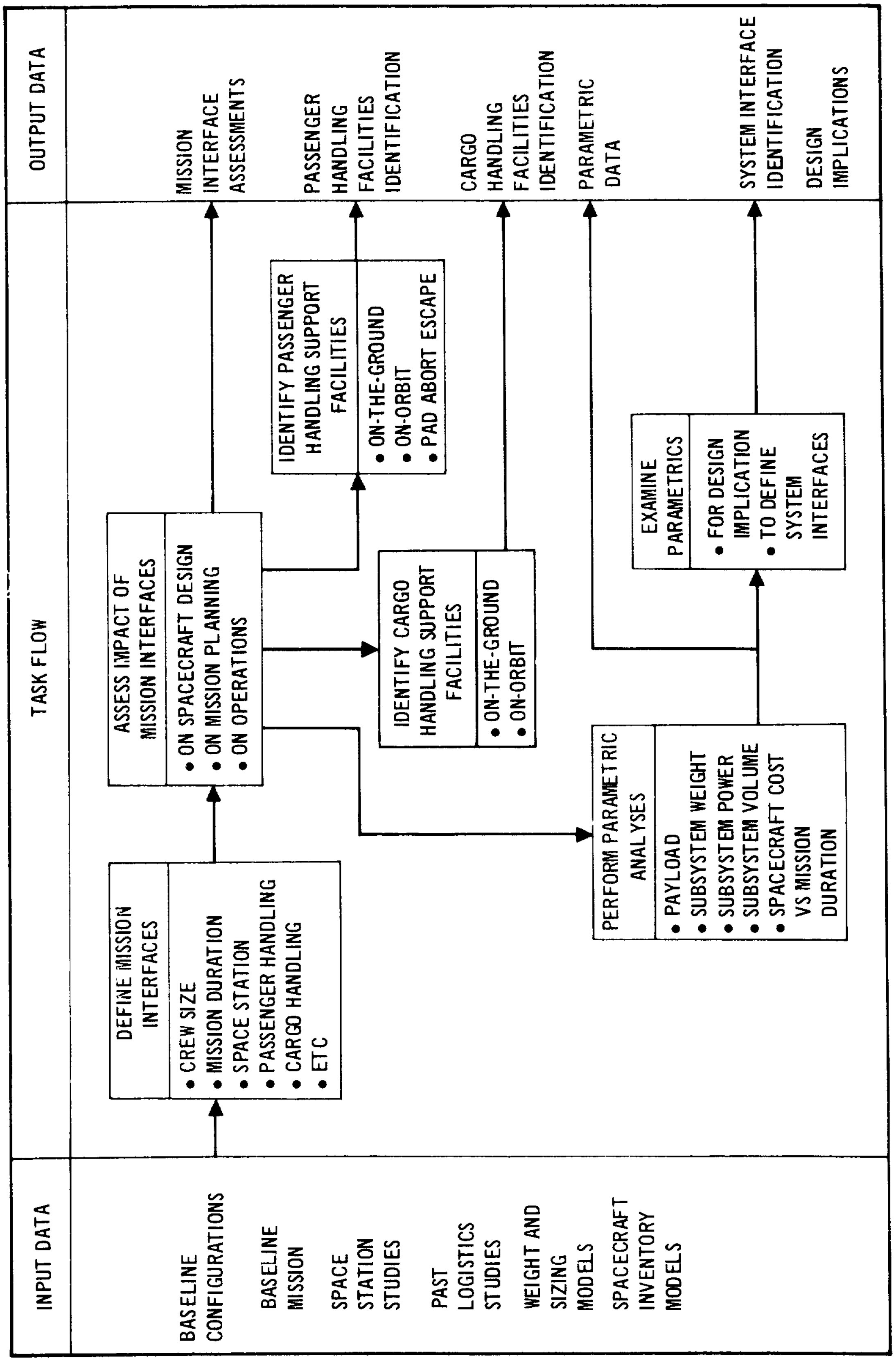

Figure 4-12

Table 4-8 MISSION INTERFACE DEFINITION

	MISSION PHASE	PROPOSED EVENT/CAPABILITY	ALTERNATE APPROACHES	REMARKS/CONSIDERATIONS
0.0	Prelaunch Operations	o Transport carrier to pad	In vertical position In horizontal position	
		o Transport orbiter to pad	In vertical position In horizontal position	
			Carrier transported first Orbiter transported first Both stages transported together	
		o Erect carrier	Horizontal launching Vertical launching	
		o Erect orbiter	Carrier erected first Orbiter erected first Both stages erected together (Mated)	
		o Mate orbiter to carrier	Back to back Belly to belly Carrier back to orbiter belly Carrier belly to orbiter back	
			On pad Off pad	
		o Load propellants	Before passenger loading After passenger loading	
			On pad Off pad	JP-4 fuel will be loaded off pad.
			LO_2 then LH_2 LH_2 then LO_2 LO_2 and LH_2 together	
			Orbiter loaded first Carrier loaded first Both stages loaded together	
		o Crew ingress on carrier	Before propellant loading After propellant loading During propellant loading Before propellant loading but egress during loading	
			Before passenger ingress After passenger ingress	
			Before mating After mating	
			Before orbiter crew loading After orbiter crew loading Load both crews together	
		o Crew ingress on orbiter	Before propellant loading After propellant loading During propellant loading Before propellant loading but egress during loading	
			Before passenger ingress After passenger ingress	
			Before mating After mating	
		o Crew size,	Unmanned One man Two men Three men	
		o Launch operations work force	1-shift operation 2-shift operation 3-shift operation	
		o Maximum on-pad time	3 days 2 days 24 hours	

Sheet 1 of 5

Table 4-8 MISSION INTERFACE DEFINITION (Continued)

MISSION PHASE	PROPOSED EVENT/CAPABILITY	ALTERNATE APPROACHES	REMARKS/CONSIDERATIONS
0.0 Prelaunch Operations (Cont.)	o Load passengers	On Pad Off pad	
		Before mating After mating	
		Before propellant loading After propellant loading	
		Upright seating Passengers lying on backs	Swivel-type seats.
	o On-pad (engines-out) abort, passengers	Egress through crew cabin Open payload doors and remove payload module Egress through escape tunnel	
	o On-pad abort escape, crew and passengers	Use single-man elevator on gantry Use multi-man elevator on gantry Slide down cables to safe area Use "Cherry Picker" to remove crew and passengers.	
	o On-pad abort, carrier, crew	No abort capability Ejection Seats Escape capsule Quick-egress hatches	For development flights only.
	o On-pad abort, orbiter, crew	No abort capability Ejection seats Escape capsule Quick-egress hatches	For development flights only.
1.0 Ascent	o Holddown at engine ignition	No holddown 1-sec. holddown 2-sec. holddown 3-sec. holddown	
	o Liftoff	Orbiter engines off Orbiter engines at full thrust Orbiter engines in idle mode	
	o Engine-out capability	None Carrier only Orbiter only Both stages	
	o Maximum acceleration during launch	3g's 4g's 5g's 6g's	With passengers aboard.
	o Low-altitude abort (before staging), carrier	No abort capability Ejection seats Escape capsule Separate & fly back	For development flights only.
	o Low-altitude abort (before staging), orbiter	No abort capability Ejection seats Escape capsule Separate & fly back	For development flights only.
	o Low-altitude abort (before staging), orbiter	Ejection Seats	Operational flights after T_o + 20 seconds
	o High-altitude abort (after staging), carrier	Intact abort capability Ejection Seats Escape capsule	For development flights only.
	o High-altitude abort (after staging), orbiter, crew	Intact abort capability Ejection Seats Escape capsule	For development flights only.
	o High-altitude abort (after staging), orbiter, passengers	Intact abort capability Ejection Seats Escape capsule Eject and recover payload canister intact	
	o Parking orbit altitude	45 x 100 NM 100 NM, circular None, Direct ascent	Low-energy, low-heat considerations.

Sheet 2 of 5

Table 4-8 MISSION INTERFACE DEFINITION (Continued)

MISSION PHASE	PROPOSED EVENT/CAPABILITY	ALTERNATE APPROACHES	REMARKS/CONSIDERATIONS
1.0 Ascent (Cont.)	o Transfer technique to space station orbit	Ground hold phasing Parking orbit phasing Rendezvous compatible orbits Limited Revolution Modified limited revolution	 Result of trade study.
2.0 Orbital Operations	o Payload Removal from cargo bay of orbiter	Accomplished by orbiter Accomplished by space station station Accomplished by space tug Accomplished by payload itself.	Via 2-way translational device Via integral RCS
		Occurs before docking Occurs after docking Occurs without orbiter docking	
		From front of orbiter via swingnose From top of orbiter From bottom of orbiter From side of orbiter From rear of orbiter via swingtail	
		Payload is single integral unit Payload is built from smaller modules	
	o Transfer payload from orbiter to space station	Accomplished by orbiter Accomplished by space station Accomplished by space tug Accomplished by payload itself	 Via integral RCS
	o Payload is docked to space station	Zero-g, non-rotating station Artificial-g station	
		Payload does docking maneuvers Space tug does docking maneuvers Space station does docking maneuvers	
		Docking is external to space station Docking is internal to space station	
		Payload is docked on end Payload is docked on side	Requires more in-depth study.
		Docking via visual sighting only Docking via electronics only Docking via combination of visual sighting and electronics	
	o Passenger transfer	Suited transfer Transfer via EVA Shirtsleeve transfer	 Via hatch at one end of payload.
	o Crew transfer	Crew does not transfer Crew transfers with passengers inside payload canister Crew transfers via separate docking	
		Return crew is same as up crew Return crew is not same as up crew Combination of above	
	o Cargo transfer	Entire cargo transferred after docking Cargo transferred on a "use" basis	 After an initial minimal transfer of special equipment
		Crew participates in cargo transfer Crew does not participate in cargo transfer	
		Passengers participate in cargo transfer Passengers do not participate in cargo transfer	 Except when absolutely necessary.

Sheet 3 of 5

Table 4–8 MISSION INTERFACE DEFINITION (Continued)

MISSION PHASE	PROPOSED EVENT/CAPABILITY	ALTERNATE APPROACHES	REMARKS/CONSIDERATIONS
2.0 Orbital Operations (Cont.)	o Orbiter status during payload transfer	All orbiter systems shutdown All orbiter systems placed on standby All orbiter systems on Orbiter under manual control, crew present Orbiter under automatic control, crew present Orbiter under automatic control, crew absent Return vehicle is same as up vehicle. Return vehicle not same as up vehicle	
	o Orbiter performs operational support to space station	Aid in station keeping Placement/retrieval of remote sensors/hardware Maintenance and repair of station All of the above Provide no operational support Perform only planned mission while on orbit Perform additional unplanned missions while on orbit	 Includes operational support.
	o Cargo loading	Just before separation and return On a gradual build-up basis	
	o Payload undocks from space station	Return payload canister same as up canister Return payload canister not same as up canister	
	o Payload is docked to orbiter	Payload does docking maneuver Orbiter does docking maneuver Space tug does docking maneuver Space station does docking maneuver	
	o Payload loading onto orbiter	Via autonomous action Via action by orbiter Via action by space tug Via action by space station	 With radio control from payload
3.0A Orbiter Descent	o Orbiter landing	Near launch site At one of various remote landing sites Manual landing Automatic landing Automatic landing with manual override	
3.0B Carrier Descent	o Carrier cruises to landing site	Inflight refueling required Sufficient onboard fuel to cruise to downrange landing site Sufficient onboard fuel to cruise back to launch site	
	o Carrier landing	Near launch site At one of various downrange landing sites Manual landing Automatic landing Automatic landing with manual override	

Sheet 4 of 5

Table 4-8 MISSION INTERFACE DEFINITION (Continued)

MISSION PHASE	PROPOSED EVENT/CAPABILITY	ALTERNATE APPROACHES	REMARKS/CONSIDERATIONS
4.0A Orbiter Maintenance	o Passenger Egress	Passengers leave S/C singly under own power	
		Passengers taken as a group intact with payload canister	Requires special equipment
	o Crew egress	Before passenger egress After passenger egress	
		Before S/C Cooldown is complete After S/C cooldown is complete	
		Crew leaves S/C unassisted Crew is taken from S/C	Via "Cherry picker"
	o Maintenance work force, Post-flight maintenance	1-shift operation 2-shift operation	
	o Maintenance work force, Pre-flight maintenance	1-shift operation 2-shift operation 3-shift operation	
	o Length of maintenance	14-day operation 10-day operation 7-day operation 3-day operation	20th operation, 90% learning
	o Prepare Orbiter for next flight	On pad Off pad	At maintenance hanger.
		Vertical assembly Horizontal assembly	
	o Load new payload canister	On pad Off pad	
		Vertical loading Horizontal loading	Less passengers
	o Load jet fuel	On pad Off pad	
4.0B Carrier Maintenance	o Crew egress	Before S/C cooldown is complete After S/C cooldown is complete	
		Crew leaves S/C unassisted Crew aided from S/C	Via "cherry picker"
	o Maintenance work force, post-flight maintenance	1-shift operation 2-shift operation	
	o Maintenance work force, pre-flight maintenance	1-shift operation 2-shift operation 3-shift operation	
	o Length of Maintenance operation	14-day operation 10-day operation 7-day operation 3-day operation	20th operation, 90% learning
	o Prepare carrier for next flight	On pad Off pad	
		Vertical assembly Horizontal assembly	
	o Load jet fuel	On pad Off pad	

Sheet 5 of 5

- o How is the payload extracted from the Orbiter?
- o How is the payload to be transferred from the Orbiter to the Space Station?
- o How is the payload to be docked with the Space Station?
- o How is the payload transferred back to the Orbiter from the Space Station?
- o How is the payload placed back into the payload bay of the Orbiter?

Additionally, two other questions related to crew and passenger transfer seem appropriate:

- o Is a crew-access tunnel from the crew cabin to the payload bay necessary or desirable?
- o Is a passenger quick-egress tunnel from the payload bay to the outside of the Orbiter necessary or desirable?

In the following paragraphs, each of the above questions is analyzed and answered as an illustration of the procedures used in identifying the mission interfaces and their impact on mission planning and/or system design.

It should be noted here that, by MDAC groundrule, the operational modes of docking the Orbiter directly to the Space Station and of docking the payload to the Space Station while it is still physically attached to the Orbiter were not considered. The rationale behind this groundrule is that the combined mass of the Orbiter plus payload is much greater than that of the Space Station, especially during the early years of the Space Station buildup, and the attendant attitude control problems incurred while docking would be prohibitively large. However, as the Space Station becomes larger and larger, through gradual buildup, this groundrule may not remain valid, and direct docking could become the preferred operational mode.

a) On-Orbit Payload Unloading - The removal of the payload from the Orbiter can be accomplished in one of three general ways:

- o By Orbiter-initiated and controlled methods
- o By payload-initiated and controlled methods
- o By methods initiated and controlled by a third vehicle, i.e., by the Space Station or by the Space Tug.

More specifically, the following list contains a number of the more feasible methods for extracting the payload from the payload bay:

- o Translational devices:
  - Telescopic pushers
  - Worm gear pushers
  - Loaded springs
  - Scissor extendors
  - Cable reel-in devices
  - Inflatable devices
- o Swing-out docking ring
- o Payload attached to payload bay door & swings out with door opening
- o Space Tug docks with payload and pulls payload out
- o Space Station uses winch, boom, or arm
- o Payload removes itself through use of propulsive devices

Representative payload-unloading concepts are pictured in Figure 4-13. An assessment of each method as to the advantages and disadvantages of its use is presented in Table 4-9. Based on this assessment, the use of two-way translational devices for on-orbit payload unloading is selected as the preferred mode. The other methods exhibit major alternate mission limitations and/or serious dynamic problems.

b) On-Orbit Payload Transfer - Once unloaded from the payload bay of the Orbiter, the payload is transferred to the Space Station. Four of the more promising ways of accomplishing this task are:

- o Use of a self-contained payload maneuvering system
- o Pushing or pulling by a Space Tug
- o Cable reel-in or boom/arm withdrawal by the Space Station
- o The Space Station comes to the payload

Each of these methods is illustrated in Figure 4-14. An assessment as to the advantages and disadvantages of each method is given in Table 4-10. As a result of this assessment, the choice narrows down to two, namely, the autonomous-payload method and the use of the Space Tug (pushing). Further study is required to make a final selection. However, for the purposes of selecting a single-path operational mission, the use of the Space Tug (pushing) is selected as the preferred system.

METHODS OF PAYLOAD UNLOADING

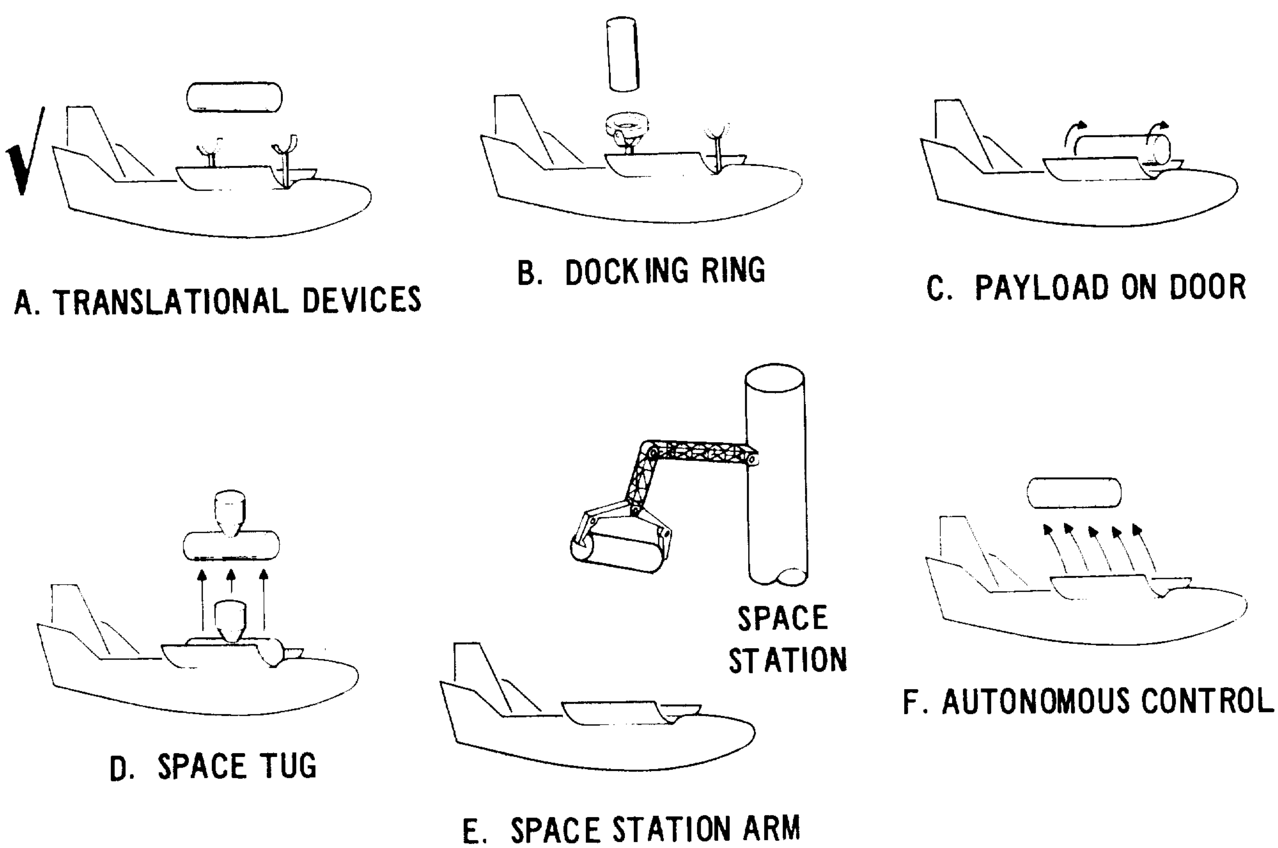

Figure 4-13

Table 4-9

PAYLOAD UNLOADING METHOD ASSESSMENT

METHOD	ADVANTAGES	DISADVANTAGES
1. Translational Devices	(a) For 2-way push and hold devices - can be used for alternate missions.	(a) Weight charged against payload. (b) Some are one-way only devices. (c) Mechanisms require volume and may interfere with system placement.
2. Swing-Out Docking Ring	(a) Can be used for alternate missions.	(a) Weight charged against payload. (b) Mechanism requires volume and may interfere with system placement. (c) Translate and swing mechanism may be exceeding large.
3. Swing-Out On Door	(a) Can be used for alternate missions. (b) Only one operation.	(a) Mechanism requires volume and may interfere with system placement. (b) Dynamic forces on door & door hinges may be excessive.
4. Space Tug	(a) Can be used for some alternate missions. (b) No internal mechanisms required.	(a) Requires third vehicle. (b) Requires docking while payload is still in payload bay. (c) Possible problem with door interference. (d) Alternate missions requiring removal of payload canister (away from station cannot be performed.
5. Space Station Arm/Boom	(a) No internal mechanisms required.	(a) Difficult payload attachment. (b) Dynamic forces on space station may be excessive. (c) Lift out stability hard to control. (d) Alternate missions requiring removal of payload canister (away from station) cannot be performed.
6. Payload Removes Self	(a) Can be used for alternate missions.	(a) Weight charged against payload. (b) Requires additional hardware on payload canister. (c) Likely plume impingement on orbiter vehicle. (d) Return to orbiter payload bay is difficult.

METHODS OF ON-ORBIT PAYLOAD TRANSFER

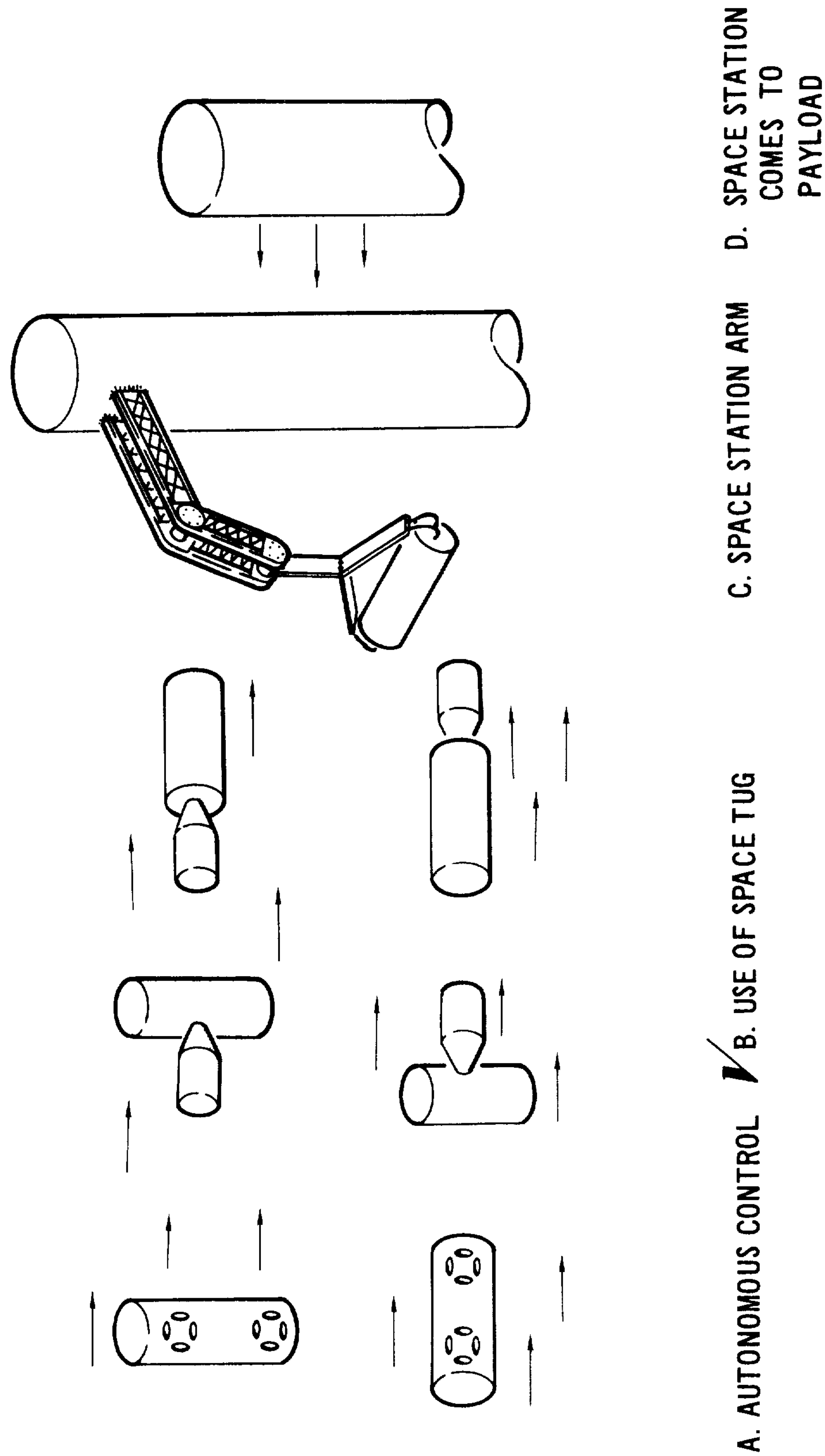

Figure 4–14

Table 4-10

ON-ORBIT PAYLOAD TRANSFER ASSESSMENT

METHOD	ADVANTAGES	DISADVANTAGES
• AUTONOMOUS CONTROL	• GOOD VISIBILITY FOR STATION DOCKING. • SIMPLE SYSTEM. • ADAPTABLE TO DIFFERENT DOCKING CONFIGURATIONS.	• MANEUVERING SYSTEM WEIGHT CHARGED AGAINST PAYLOAD.
• USE OF SPACE TUG (PUSHING)	• DOCKING MECHANISMS ONLY ADDITIONAL HARDWARE REQUIRED. • SIMPLE SYSTEM IF SPACE TUG ALREADY EXISTING. • ADAPTABLE TO DIFFERENT DOCKING CONFIGURATIONS.	• REQUIRES USE OF THIRD VEHICLE. • POOR VISIBILITY FOR DOCKING TO SPACE STATION.
• USE OF SPACE TUG (PULLING)	• DOCKING MECHANISMS ONLY ADDITIONAL HARDWARE REQUIRED, • SIMPLE SYSTEM IF SPACE TUG ALREADY EXISTING. • GOOD VISIBILITY FOR STATION DOCKING	• REQUIRES USE OF THIRD VEHICLE. • LIMITED DOCKING CONFIGURATIONS. • REQUIRES SEPARATE SPACE TUG FOR EACH PAYLOAD CANISTER. • SPACE TUG MUST HAVE GO-THROUGH PRESSURIZED TUNNEL.
• SPACE STATION CABLE ARM/BOOM WITHDRAWAL	• GOOD VISIBILITY FOR PAYLOAD ATTACHMENT TO STATION. • VERY LITTLE ADDITIONAL HARDWARE REQUIRED ON PAYLOAD.	• INTRODUCES LARGE DYNAMIC FORCES ON SPACE STATION. • ORIENTATION OF PAYLOAD DIFFICULT TO CONTROL. • ATTACHMENT OF PAYLOAD TO STATION DIFFICULT TO ACCOMPLISH.
• SPACE STATION COMES TO PAYLOAD	• SIMPLE SYSTEM • GOOD VISIBILITY FOR STATION-TO-PAYLOAD DOCKING. • DOCKING MECHANISMS ONLY ADDITIONAL HARDWARE REQUIRED.	• MANEUVERING PROPELLANT REQUIREMENT EXCESSIVE, PARTICULARLY AS STATION BUILDUP CONTINUES.

The use of the Space Tug is further illustrated in Figure 4-15. Note that the payload is translated out from the Orbiter to clear the doors before docking by the Space Tug. This is accomplished by two-way translational device as defined in Section 4.2.2a.

c) Payload Docking - There are many docking configurations by which the payload can be attached to the Space Station. Eleven different docking configurations have been identified and are listed below:

- o Payload end to Station side
- o Payload side to Station side
- o Payload side nested in Station side
- o Payload end nested in Station side
- o Payload extended through Station
- o Payload end inserted into Station side hatch
- o Payload side to Station end
- o Payload side nested in Station end
- o Payload taken in through Station end
- o Payload taken in through Station side
- o Payload end to Station end

Each of these configurations is illustrated in Figure 4-16. An assessment of each configuration is given in Table 4-11. On the basis of these assessments and using a minimal payload-Station interface as the prime criterion, Configurations 1, 2, 7, and 11 appear to be the more promising of the group. Of these, Configuration 1, payload end to station side, is selected as the reference docking configuration only to provide operational continuity.

d) On-Orbit Return Transfer - The methods for the transfer of the payload from the Station to the Orbiter are similar to those required for the reverse transfer situation. Thus, use of the Space Tug is selected as the preferred method for return payload transfer.

e) On-Orbit Payload Loading - The loading of the payload back onto the Orbiter for return from orbit is somewhat different than its unloading. Here, the payload must be pushed or pulled into the cargo bay. Devices such as the loaded springs, cable reel-in motors, and inflatable devices are usually one-way expulsion mechanisms and can therefore be dropped from any further consideration.

PAYLOAD TRANSFER SEQUENCE

Space Tug

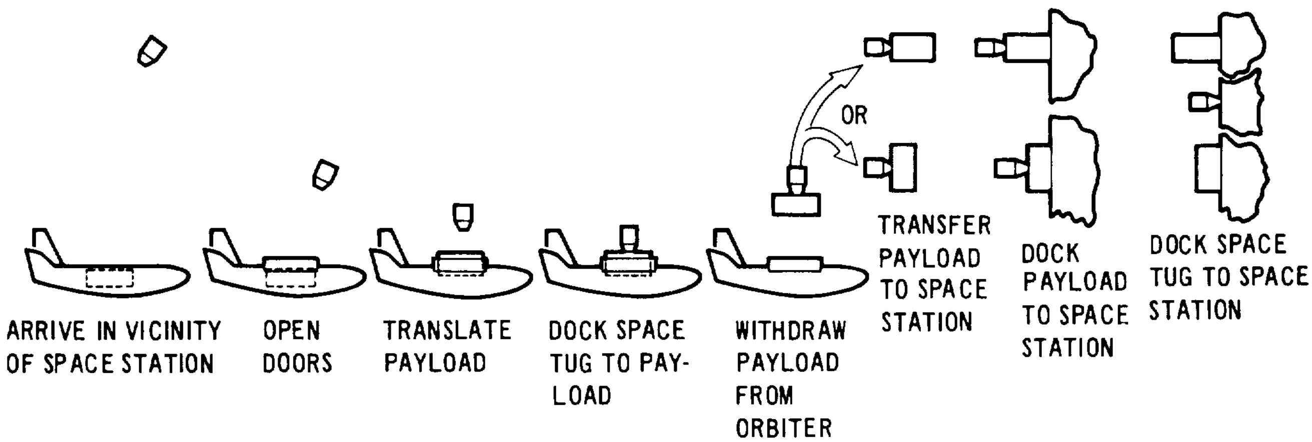

Figure 4-15

METHODS OF DOCKING PAYLOAD TO SPACE STATION

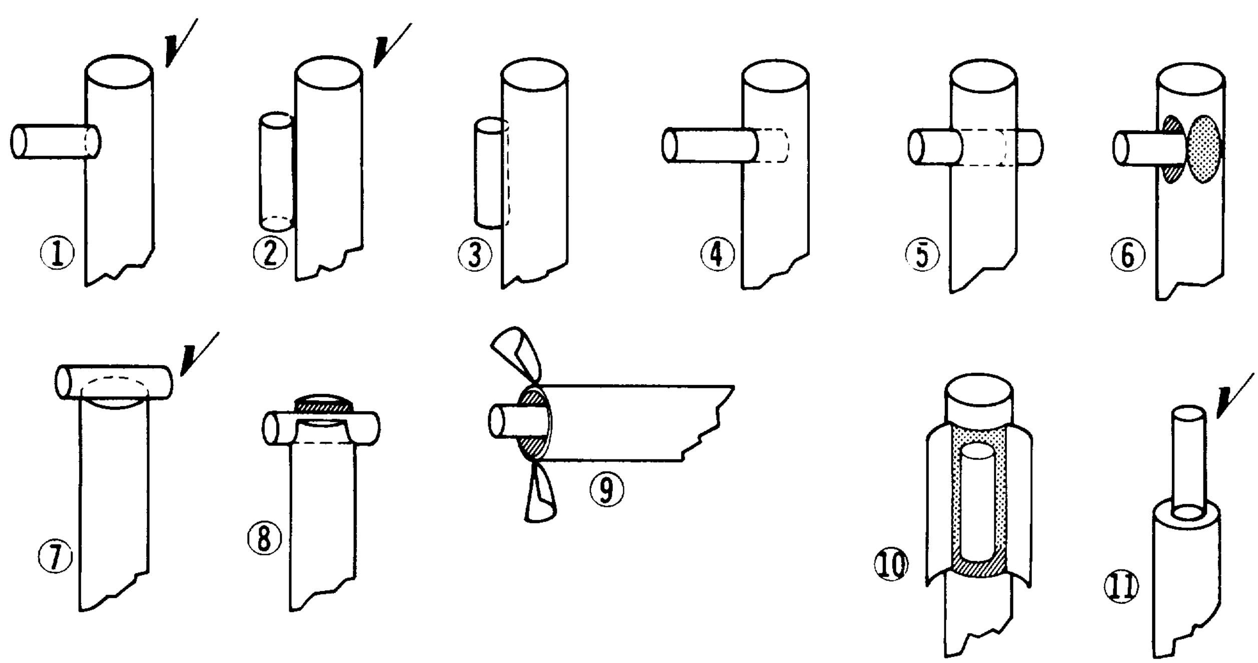

Figure 4-16

Table 4-11

DOCKING CONFIGURATION ASSESSMENT

CONFIGURATION	ADVANTAGES	DISADVANTAGES
1. Payload End to Station Side	(a) Simple. (b) Minimal payload - Station interface	(a) Some attitude control problems introduced to Station.
2. Payload Side to Station Side	(a) Simple. (b) Small payload - Station interface	(a) Some attitude control problems introduced to Station (less than 1).
3. Payload Side Nested in Station Side	(a) More contact surface, thus cargo/passenger transfer is easier.	(a) Some attitude control problems introduced to Station (less than 2). (b) Large payload - Station interface. (c) Docking is more difficult than 2.
4. Payload End Nested in Station Side	(a) More contact surface, thus cargo/passenger transfer is easier (less than 3).	(a) Some attitude control problems introduced to Station (more than 3). (b) Large payload - Station interface. (c) Docking is more difficult than 1.
5. Payload Extended Through Station	(a) More area available for cargo/passenger transfer.	(a) Some attitude control problems introduced to Station (less than 4). (b) Large payload - Station interface. (c) Docking is extremely difficult. (d) Sealing of Station openings may be difficult.
6. Payload Inserted into Station Side Hatch	(a) More area available for cargo/passenger transfer.	(a) Some attitude control problems introduced to Station (about same as 4). (b) Large payload - Station interface. (c) Docking is more difficult than 1. (d) Sealing of Station openings may be difficult.
7. Payload Side to Station End	(a) Simple. (b) Minimal payload - Station interface.	(a) Some attitude control problems introduced to Station (less than 1).
8. Payload Side Nested in Station End	(a) More area available for cargo/passenger transfer.	(a) Some attitude control problems introduced to Station (less than 7). (b) Large payload - Station interface. (c) Docking is more difficult than 7. (d) Sealing of Station openings may be difficult.
9. Payload Inserted into Station End	(a) Maximum area available for cargo/passenger transfer.	(a) Some attitude control problems introduced to Station. (b) Large payload - Station interface. (c) Docking is more difficult than 1.
10. Payload Taken in Through Station Side.	(a) Maximum area available for cargo/passenger transfer.	(a) Some attitude control problems introduced to Station. (b) Large payload - Station interface. (c) Docking is more difficult than 2.
11. Payload End to Station End	(a) Simple. (b) Minimal payload - Station interface. (c) More contact surface than 1.	(a) Some attitude control problems introduced to Station.

In addition, payload hookup and reel-in by the Orbiter is a method which should receive some consideration. Orbiter-located pinchers or grabbers (holders) are also feasible concepts. Docking to the payload by the Orbiter is another alternative worth consideration.

On the basis of having the simplest system for the entire on-orbit mission, payload grabbers were selected as the preferred system for payload loading. These are installed on the translational device for unloading the payload from the Orbiter. The device locks onto the payload and the payload is translated back into the payload bay of the Orbiter, whereupon the outer access doors are closed.

Note that the translate-and-hold devices also provide improved alternate-mission capability in that the payload can be extended from the Orbiter.

f) Crew Access Tunnel Assessment - The desirability of incorporating a crew cabin-to-payload access tunnel into the design of the Orbiter is seen when the advantages of such a tunnel are weighed against its disadvantages. Both are listed in Table 4-12. The most important of the advantages of such a tunnel is that it gives the crew access to the payload bay while on orbit, thus providing the vehicle with increased alternate mission capability and allows transfer of the crew to the station internally within the payload module.

The placement of the crew-access tunnel is pictured in Figure 4-17. As shown, the crew-access tunnel connects the crew cabin to the payload bay, running along the top and down the center of the Orbiter. Further detail as to the design of the tunnel and its interaction with other Orbiter systems is beyond the scope of this special emphasis study.

Table 4-12

CREW-ACCESS TUNNEL ASSESSMENT

ADVANTAGES
- CREW HAS ACCESS TO CARGO FOR ON-ORBIT OPERATIONS AND ALTERNATE MISSION CAPABILITY
- PROVIDES CREW-TRANSFER-TO-SPACE STATION CAPABILITY VIA CARGO MODULE
- POSSIBLE ALTERNATIVE ESCAPE ROUTE DURING ABORT SITUATIONS

DISADVANTAGES
- REQUIRES ADDITIONAL PRESSURIZATION AND POWER
- MAY INTERFERE WITH PROPELLANT TANK PLACEMENT
- MAY REQUIRE PLACEMENT OUTSIDE OF ORBITER MOLDLINE
- USES VOLUME OTHERWISE AVAILABLE FOR ORBITER SYSTEMS

CREW-ACCESS TUNNEL PLACEMENT

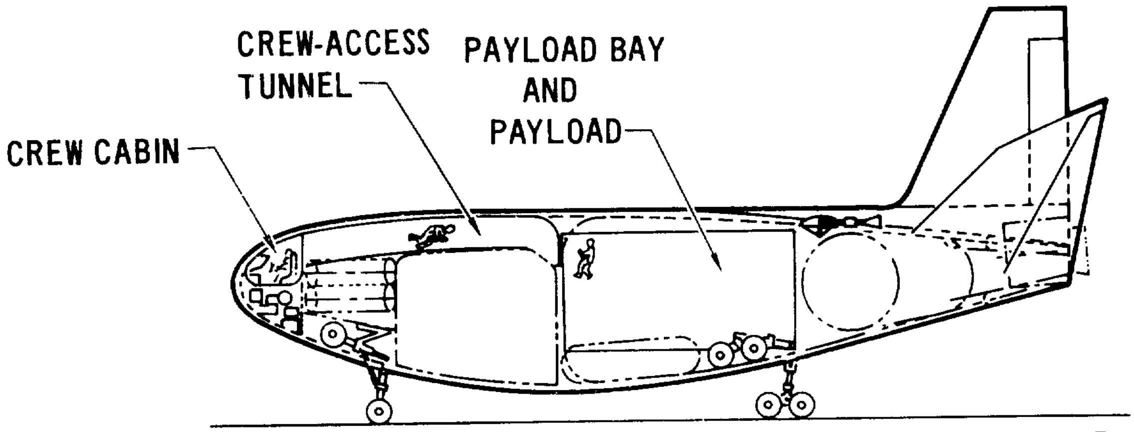

Figure 4-17

g) Passenger On-Pad Quick Egress - With as many as ten passengers scheduled to travel within the payload canister onboard the Orbiter, there is little doubt that some method of passenger quick egress should be provided for on-pad emergency situations. Five quick-egress procedures for passenger escape are suggested in Table 4-13. The advantages and disadvantages of each of these procedures are listed in Table 4-14. Of the five alternatives, the method of the quick egress tunnel was adjudged to be the simplest and quickest. However, the adequacy of the technique must be determined by further study.

Table 4-13

METHODS OF ON-PAD PASSENGER QUICK EGRESS

- BLOW HATCH IN PAYLOAD CANISTER, TRAVERSE TUNNEL, OPEN HATCH IN PAYLOAD DOORS, SLIDE DOWN CABLE
- OPEN PAYLOAD DOORS, TRANSLATE PAYLOAD CANISTER OUT, OPEN HATCH IN PAYLOAD CANISTER, SLIDE DOWN CABLE
- OPEN PAYLOAD DOORS, REMOVE PAYLOAD CANISTER, TRANSFER ENTIRE PAYLOAD CANISTER TO SAFE AREA
- OPEN HATCH IN PAYLOAD CANISTER, CLIMB THROUGH CREW-ACCESS TUNNEL, ESCAPE THROUGH CREW QUICK-EGRESS HATCHES, SLIDE DOWN CABLE
- BLOW HATCH IN PAYLOAD CANISTER, OPEN PAYLOAD DOORS, SLIDE DOWN CABLE

Table 4-14

PASSENGER ON-PAD QUICK EGRESS METHOD ASSESSMENT

Method	Advantages	Disadvantages
1. Escape Tunnel	o Simple o Very little additional equipment o Minimal physical effort required of passengers	o Slow (probably fastest method)
2. Utilization of Onboard Translational Devices	o Simple o Uses existing equipment o Minimal physical effort required of passengers	o Relatively slow (slower than 1)
3. Remove Payload Canister Intact	o No physical effort required of passengers	o Requires heavy equipment o Very slow o Complex
4. Utilize Crew-Access Tunnel	o Simple o Uses existing equipment	o Prohibitively slow o Requires large physical effort by passengers
5. No Escape Tunnel, Open Doors	o Simple o Uses existing equipment o Minimal physical effort required of passengers	o Relatively slow (slower than 1) o Nothing to bridge gap between payload and doors

On the basis that a passenger quick-egress tunnel is the preferred method of extracting the passengers in an emergency situation, an assessment as to the desirability of incorporating such a tunnel into the baseline design was made. The advantages and disadvantages of the employment of a quick egress tunnel are listed in Table 4-15. With these in mind, and with a view to the not-too-distant past Apollo tragedy, it was decided to include the escape tunnel in the baseline vehicle design.

The passenger quick-egress tunnel is envisioned as a non-pressurized tunnel physically attached to one of the payload bay doors. At the payload bay door interface there would be a smaller door, operable from the inside. At the payload end of the tunnel, i.e., at the payload canister-tunnel interface, a quick-opening hatch would be provided. The tunnel would be short, extending only from the payload canister to the payload bay doors and may be in an inclined position when the Orbiter is vertical to allow quicker passage of the passengers.

Table 4-15

QUICK-EGRESS TUNNEL ASSESSMENT

ADVANTAGES
- PROVIDES ON-PAD QUICK EGRESS FOR PASSENGERS DURING ENGINES – DOWN ABORT
- PROVIDES ALTERNATE ON-PAD EGRESS FOR CREW DURING ENGINES – DOWN ABORT
- PROVIDES ON-PAD PASSENGER INGRESS WITH PAYLOAD DOORS CLOSED

DISADVANTAGES
- USES VOLUME OTHERWISE AVAILABLE FOR ORBITER SYSTEMS
- MAY INTERFERE WITH PROPELLANT TANK PLACEMENT

4.2.3 Payload-Handling Facilities - For discussion purposes, payload-handling facilities can generally be broken down into two categories, on-the-ground facilities and on-orbit facilities. Ground facilities can further be subdivided by geographical location, e.g., at the maintenance hangar, on the launch pad, and at the landing site, both primary and secondary. A detailed description of the ground facilities can be found elsewhere in this report (see Section 1.3, Vol. III, Facilities Plan).

On-orbit payload-handling facilities are not described in detail as this would have entailed investigations beyond the scope of the present study. However, certain major items of payload-handling equipment can be identified. For example, for payload unloading, two-way translational and holding devices would have to be provided. For intact payload transfer from the Orbiter to the Space Station, use of a Space Tug has been proposed as the preferred mode of operation. Docking hardware on the payload canister to enable its attachment to the Station would have to be provided.

Having attached the payload to the Station, and having transferred both crew and passengers, some means must be provided for transferring the cargo. It is proposed that the cargo be transferred on a "use" basis, which would eliminate most of the cargo-handling problems. However, certain general items, such as hand rails, quick-connect/disconnect tie-down devices, color-coding and modularization of the cargo, etc. would have to be provided in any case. In addition, loading and unloading would follow a pre-flight constructed cargo-transfer plan.

While the transfer of the cargo is to take place according to the rate at which the provisions, equipment, etc. are needed, the requirement to transfer certain large heavier items must also be anticipated. For items such as these, manually operated moving equipment will most likely be required. Such equipment would be located onboard the Station to enable its recurring use with numerous missions and docked payloads and in the loading of return cargo.

4.2.4 <u>Passenger-Accommodation Facilities</u> - Like the payload-handling facilities, a more detailed discussion of the ground passenger-accommodation facilities can be found in the Facilities Plan, Section 1.0, Vol. III. Aside from facilities for housing and transporting the passengers to the launch site, special provisions will have to be made for on-the-pad loading, on-pad passenger quick-egress during an emergency situation, passenger seating awaiting liftoff, environmental control and working-volume allocation during the mission, and for transferring the passengers to and from the Space Station.

Since it is anticipated that in the Operational Phase of the Space Shuttle Program, the passengers will consist of scientists, engineers, and technicians whose ages and physical fitness do not compare with that of present-day astronauts, special accommodations will have to be provided for them above what is done today. For instance, a maximum 3-g acceleration during the boost phase is decreed when passengers are aboard, whereas 4g's are allowed otherwise.

While loading the Orbiter, it is anticipated that a walk-on capability from the service tower will be available. On-pad quick-egress during an engines-down or other emergency situation is thus provided. Passenger last-minute loading and upright seating while waiting for liftoff (with swivel-seat adjustment to the on-back position just prior to ignition) will govern the prelaunch scheduling. Inside the payload canister, sufficient volume must be given to the passengers for moving around. Environmental control must be supplied since a shirtsleeve environment is a mission requirement.

On-orbit activity of the passengers aboard the Orbiter will be kept minimal and their participation in cargo transfer will not be required.

4.2.5 Mission Duration Parametrics - The baseline mission is a Space Station logistics mission in which a two-man crew pilots the orbiter with 10 passengers riding in the payload canister. The baseline mission duration is "up to 7 days". The parametric curves presented in this section show the effects (on the payload and on program costs) of increasing the mission from seven to thirty days. In addition, these effects are shown for the case where only a two-man crew is present.

a) Effect of Mission Duration on Orbiter Subsystems - The effects of increasing the mission duration from 7 to 30 days on the weight and volume of the basic Orbiter subsystems are shown in Figures 4-18 and 4-19, respectively. These two charts also illustrate the differences resulting from the assumption of a 2-man versus 12-man crew (2 crewmen plus 10 passengers).

It is seen that four Orbiter subsystems are primarily affected by the increase in mission duration, namely, attitude control (ACS), environmental control (ECS), power, and crew provisions (i.e., food and water). By weight and by volume, the ACS is the subsystem exhibiting the greatest effects of mission duration.

The effects of a 2-man crew versus a 12-man crew are most seen on the environmental control system. This effect is at its greatest for the longest duration missions. The basic power and attitude control subsystems for the Orbiter were sized for a 12-man complement and, consequently, show little effects when the crew size is increased from two to twelve men.

All increases in the Orbiter subsystems weights shown in these two charts are due to increased requirements for propellants, gases, and reactants. The crew-provisions increase in the only exception to this. Hence, except for tankage, no increase in subsystem hardware is necessary for missions of duration up to 30 days.

b) Effect of Mission Duration on Orbiter Payload - The overall weight and volume effects of increasing the length of the mission from 7 to 30 days are seen in Figures 4-20 and 4-21, respectively. Here, the increases in weight and volume of the Orbiter subsystems are charged against the baseline 25,000-pound, 5300-cu. ft. payload. Also shown are the payload differences resulting from a 2-man versus a 12-man crew, and the combined effects of both variables.

EFFECT OF MISSION DURATION ON SUBSYSTEM WEIGHT

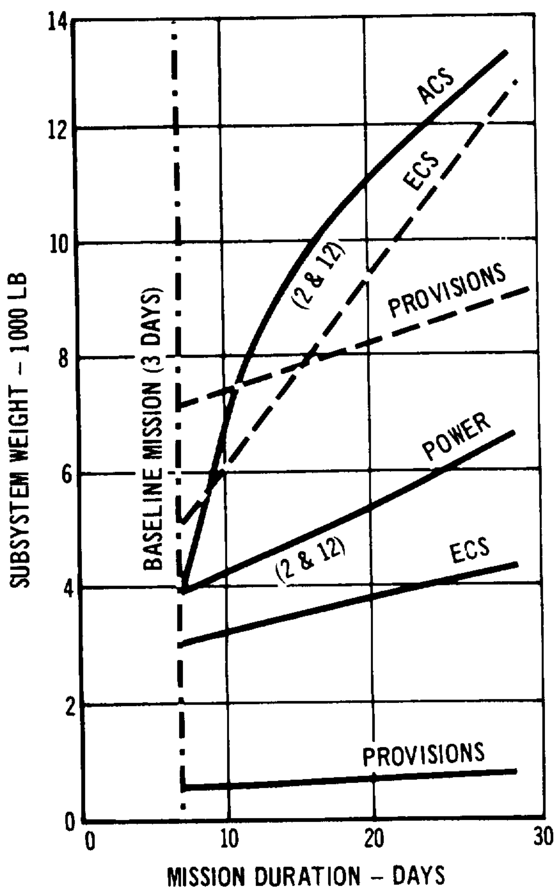

Figure 4–18

EFFECT OF MISSION DURATION ON ORBITER SUBSYSTEM VOLUME

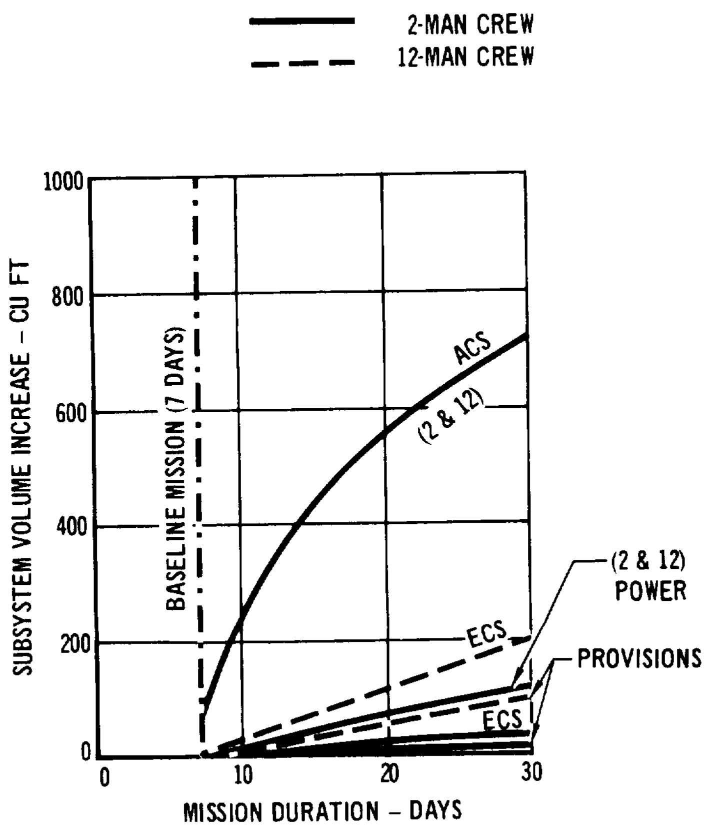

Figure 4–19

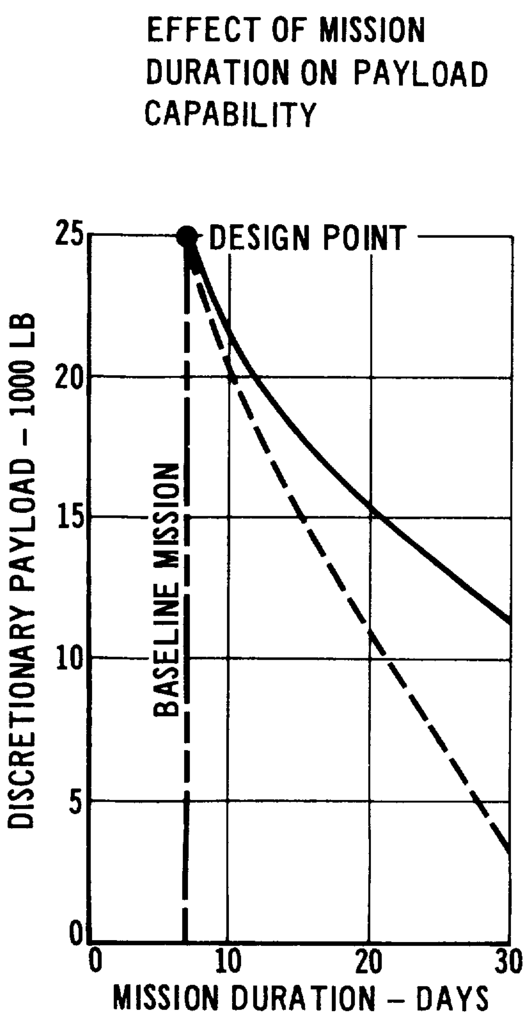

Figure 4–20

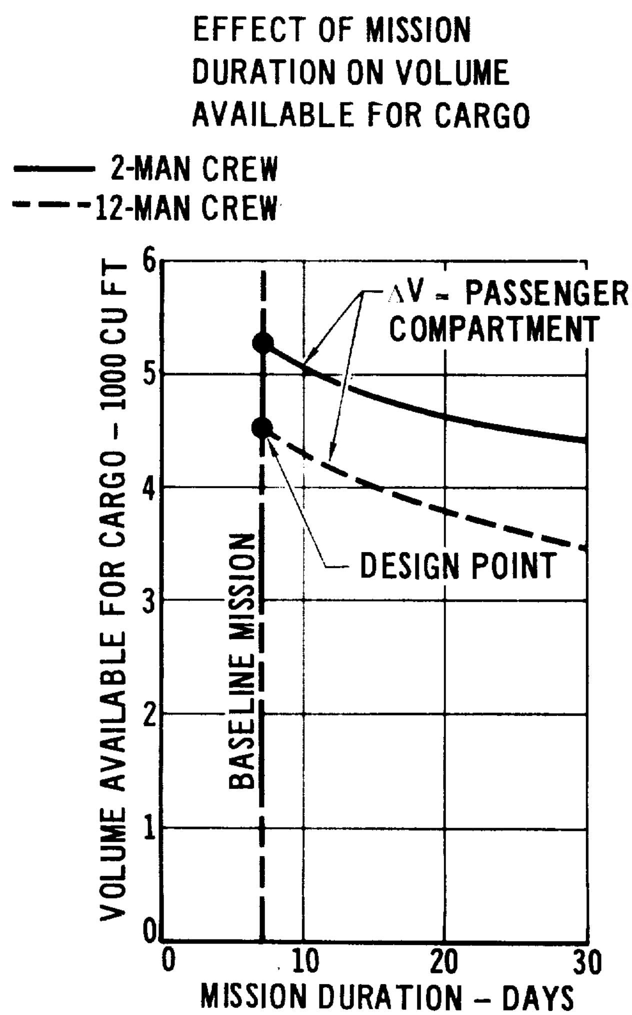

Figure 4–21

It is seen that the available Orbiter payload erodes very quickly with an increase in mission duration. In particular, a decrease from 25,000 lbs. to 11,400 lbs. results for the 2-man mission when going from 7 to 30 days. Similarly, the 12-man crew mission sees a payload decrease from 25,000 lbs. to 3400 lbs. over the same range of mission length.

In the case of the payload volume, the decrease in available volume is not nearly as severe as was the weight, with the 12-man-crew mission showing somewhat greater effects. However, in neither case does the payload volume diminish sufficiently to warrant curtailment of the mission duration on this basis alone. The volume difference between 2 and 12-man vehicles is almost entirely that resulting from the housing requirement of the additional ten men.

c) Effect of Mission Duration on Orbiter Electrical Power - The increase in Orbiter electrical energy requirements with an increase in mission duration is shown in Figure 4-22. The linear relationship stems from the assumption of a constant 110-kilowatt-hour basic daily requirement for the operational mission. Imposed on top of these requirements is the assumption of a 200-kilowatt-hour/day/man requirement for the operation of experiments (one experiment per crewman, average power of 100 watts, run for 4 hours per day). This latter requirement accounts for the difference in energy requirement between the 2-man and the 12-man missions.

d) Effects of Mission Duration on Program Costs - It is shown in Section 4.2.5a that an increase in mission duration resulted in increased Orbiter subsystem weights and volumes. Since these increases were in the form of propellants, gases, and fuel cell reactants, little increase in program costs would be seen. However, if it is assumed that all missions of the program have the same mission duration, then the basic spacecraft inventory is affected and, consequently, program recurring costs.

The basic spacecraft inventories for the missions being considered in this study are shown in Table 4-16. Both the effect of increasing the mission duration from 7 to 30 days and the nominal (no-loss) annual launch rate from 4 to 100 launches per year are shown.

EFFECT OF MISSION DURATION ON ORBITER ELECTRICAL POWER REQUIREMENTS

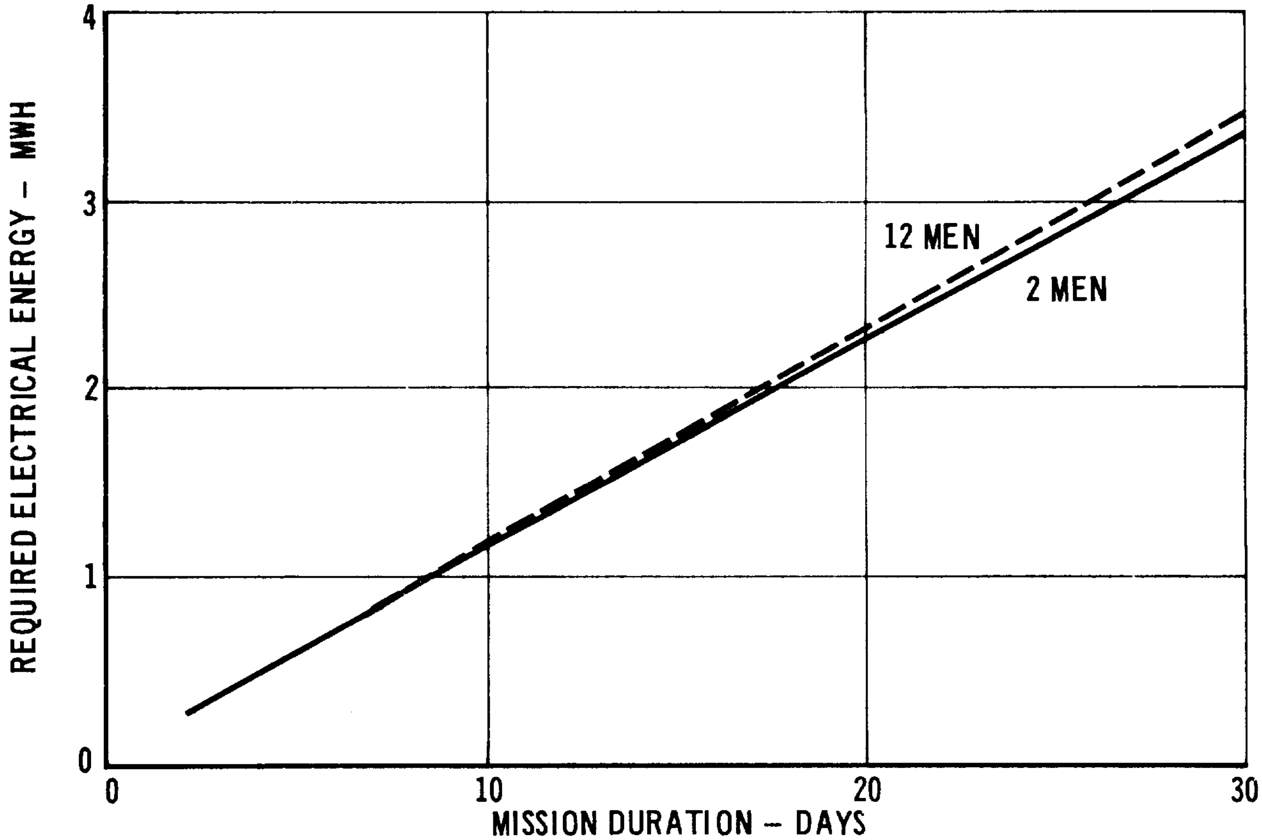

Figure 4-22

Table 4-16

SPACECRAFT INVENTORY

- 10-YEAR PROGRAM
- DESIGN LIFE = 100 USES
- 1-DAY CARRIER MISSION
- ASCENT RELIABILITY 0.97
- P_{SR} (ORBITER) = 0.990
- P_{SR} (CARRIER) = 0.995

ANNUAL LAUNCH RATE	NUMBER OF VEHICLES REQUIRED		
	CARRIER	ORBITER	
		7-DAY MISSION	30-DAY MISSION
4	2	2	2
8	2	2	2
10	2	2	3
12 REFERENCE	2	3	3
30	4	6	7
50	7	10	11
100	15	19	21

P_{SR} = PROBABILITY OF SUCCESSFUL RECOVERY

For the 10-year program, with the probabilities, reliabilities, and design life as stated on the chart, the Carrier inventory varies from 2 vehicles at the low (4 launches/year) launch rate to 15 vehicles at the high (100 launches/year) launch rate. At the same time, the Orbiter inventory varies from 2 to 19 vehicles over the same range of launch rates and for the nominal 7-day mission. For a 30-day mission, the upper limit is extended to 21 vehicles for the Orbiter.

The increase in spacecraft (Orbiter) inventory with mission duration is shown in graphic form in Figure 4-23. Since the Carrier's "mission" does not increase, no change in its inventory occurs.

As stated above, all missions in the program are assumed to have the same length. Mixes of different mission durations were not investigated. However, it is seen that for a mission of 27 days or less, no increase in the Orbiter inventory occurs, even for launch rates of 100 launches/year (1000 successful missions).

It should be noted that the spacecraft inventories are very sensitive to the inputs of design life, mission reliability, launch-into-orbit reliability, etc. Sensitivity plots of the stage inventory versus these variables are shown in Volume III, Section 2.0.

The recurring cost increases resulting from the increase in Orbiter inventory versus mission duration are shown in Figure 4-24. Since an increase in mission duration can be accomplished without a corresponding increase in the Orbiter's subsystem hardware (save for a few propellant, gas, and reactant tanks), no increase occurs in the spacecraft basic unit cost. Thus all cost increases shown are the result of increased inventories, which in turn, are the result of increased mission length.

The increase in recurring costs are less than $120 million for any of the programs considered. No cost increases occur for the 40, 80, or 120-successful-mission programs, and none occur for missions less than 27 days regardless of launch rate (for the range investigated).

EFFECT OF MISSION DURATION ON SPACECRAFT INVENTORY (ORBITER ONLY)

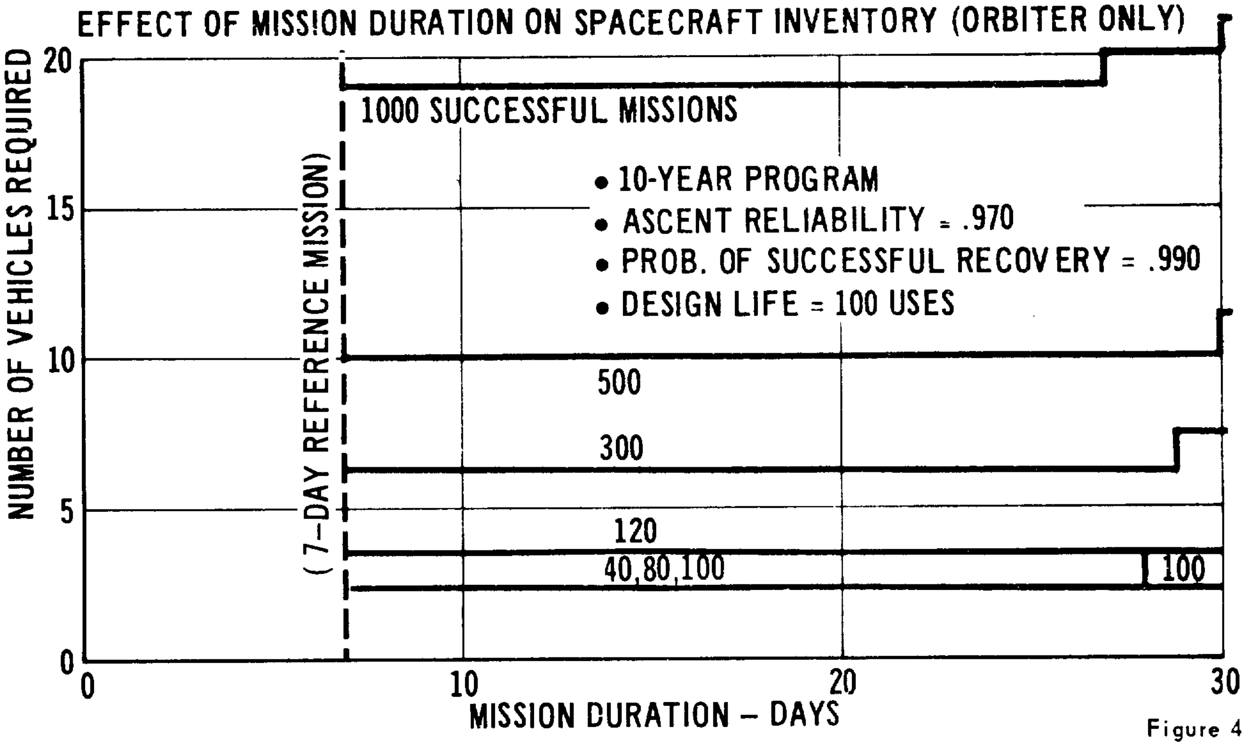

Figure 4–23

EFFECT OF MISSION DURATION ON RECURRING COSTS

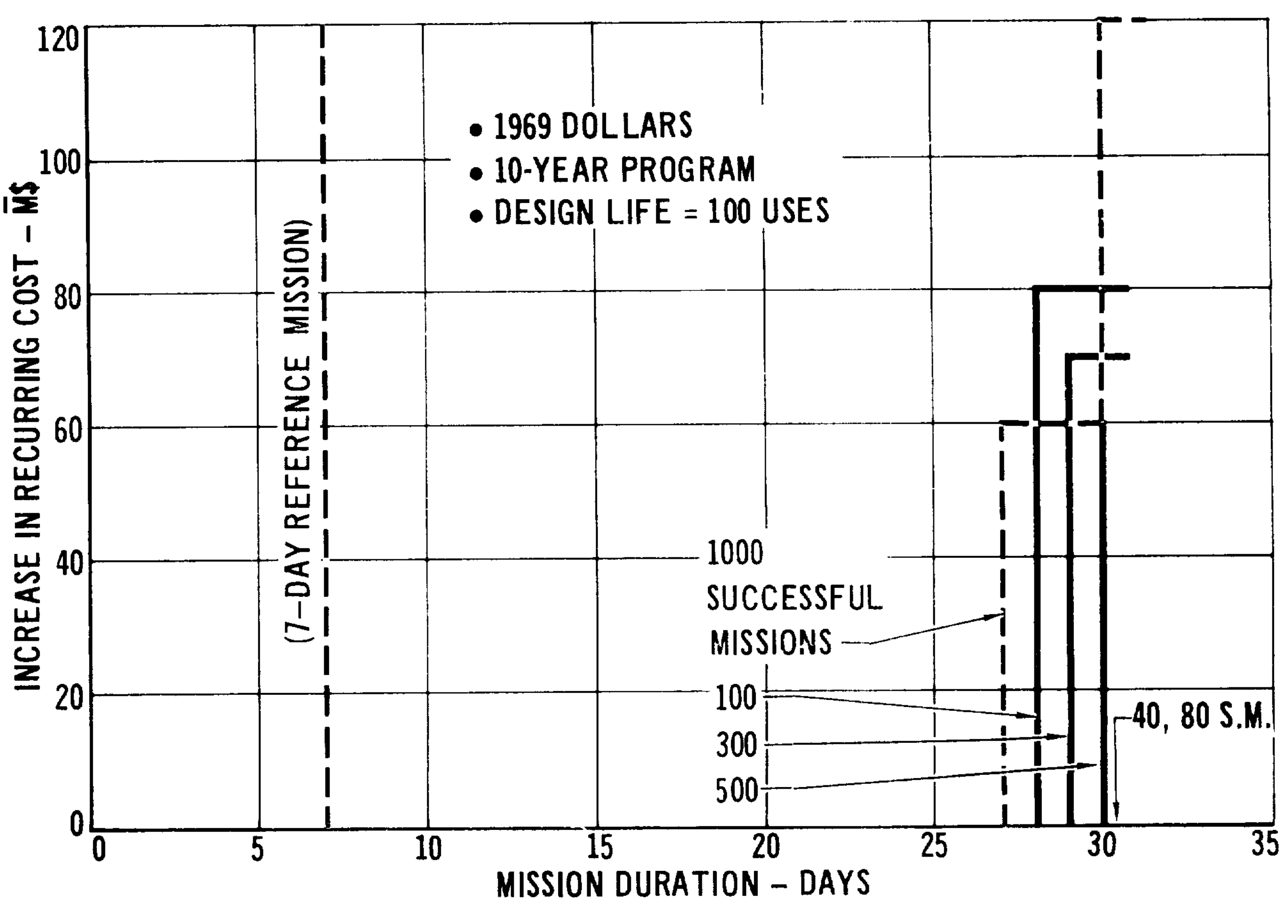

Figure 4–24

Table 4-17

SYMBOLS AND ABBREVIATIONS

ITEM	DEFINITION
ACS	Attitude Control System
AEF	After Every Flight
AGE	Aerospace Ground Equipment
APU	Auxiliary Power Unit
AVE	Aerospace Vehicle Equipment
CONUS	Continental United States
ECS	Environmental Control System
EC/LSS	Environmental Control and Life Support System
ECM	Electromagnetic Capability
ETR	Eastern Test Range
GSE	Ground Support Equipment
ILRV-LRC	Integral Launch Reentry Vehicle - Langley
JP	Jet Petroleum
KSC	Kennedy Space Center
LiOH	Lithium Hydroxide
LRU	Line Replaceable Unit
MDAC	McDonnell Douglas Astronautics Company
OBC	Onboard Checkout
OMACS	Orbital Maneuver Attitude Control System
PMEL	Precision Measurement Equipment Laboratory
PTL	Prior to Launch
QA	Quality Assurance
QC	Quality Control
S/C	Spacecraft
SM	Scheduled Maintenance
TAT	Turnaround Time
VAB	Vehicle Assembly Building

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